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FINAL REPORT

BUOYANT VENUS STATION MISSION FEASIBILITY STUDY FOR 1972 - 1973 LAUNCH OPPORTUNITIES

VOLUME III: CONFIGURATION DEFINITION
PART I: CONFIGURATION

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for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

The use of a buoyant station to explore the atmosphere of Venus has shown great promise in previous studies.* The high temperatures and pressures likely to be encountered and the unknown surface characteristics tend to make the in situ exploration of this planet difficult. The buoyant station concept permits the mission's instruments to be floated in the atmosphere in a moderate environment, localizing the environment problem to relatively compact drop sondes, and avoiding the problems of landing, deployment, and survival on the surface. Relatively simple missions can thus be defined that have the advantages of long duration and mobility over the surface, depending on the existing wind patterns. From such a station, measurements relating both to the atmosphere and the surface can be made. The surface measurements can be obtained either indirectly or by sondes dropped from the parent buoyant station.

This final report on the Buoyant Venus Station Mission Feasibility Study for 1972 - 1973 Launch Opportunities is submitted by the Martin Marietta Corporation, Denver Division, in accordance with Contract NAS1-7590.

This report is submitted in three volumes as follows:

Volume I - Mission Summary Definition and Comparison;

Volume II - Trajectory Analysis for 1972 and 1973 Missions;

Volume III - Configuration Definition.

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*J. F. Baxter: Final Report, Buoyant Venus Station Feasibility Study, Volume I, Summary and Problem Identification.

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BUOYANT VENUS STATION MISSION FEASIBILITY STUDY FOR
1972 - 1973 LAUNCH OPPORTUNITIES

VOLUME III: CONFIGURATION DEFINITION

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SUMMARY

The Buoyant Station/Entry System is investigated to determine its compatibility with specified missions. The missions considered are:

1) 1972 mission with flyby spacecraft;
2) 1973 mission with orbiter spacecraft;
3) 1973 mission in conjunction with a Mercury/Venus swingby opportunity.

A baseline capsule system -- aeroshell, buoyant station, and subsonic probe -- is defined. There are many points of similarity for all three missions.

The configurations and the subsystem concepts are described to the depth necessary to identify major problem areas and to make comparisons between missions. Development of a balloon system, while a candidate for early effort, appears to be well within present technology. The heat shield development will, of course, be common for all Venus Probes.

INTRODUCTION

The objective of the study was to investigate the feasibility of several specific Venus exploration missions using the buoyant station concept. Trajectory analyses were performed -- interplanetary, orbital, flyby approach, and atmospheric entry -- by defining modifications to specified interplanetary spacecraft and by defining the configuration of the buoyant station entry system. The final steps of the study consisted of comparing the specified missions, including preparation of preliminary cost and schedule information.

The technical guidelines to which the study was performed are as follows:

1) The missions to be investigated and compared are -

    1972 flyby mission,

    1973 orbiter mission,

    1973 Mercury/Venus swingby;

2) The SLV 3C/Centaur and Titan IIIC will be considered as candidate launch vehicles;

3) Interplanetary trajectory parameters, as defined in JPL TM 33-334, TM 33-342, and TM 32-1062 shall be used;

4) Venus atmospheres are as defined in NASA SP-3016;

5) A complete DSIF network will be assumed to be available;

6) The Mariner 1969 as defined in Mariner Mars 1969 Functional Requirements shall be used for the flyby mission spacecraft;

7) The modified Boeing Lunar Orbiter as defined in NASA CR-66302 will be used for the orbital mission spacecraft;

8) The Mercury/Venus spacecraft will be as defined in JPL document 760-1;

9) The spacecraft will not be sterile;

10) The spacecraft may serve as relay stations for data transmission;

11) The minimum weight BVS (investigated under Contract NAS1-6607) will be used as the baseline station configuration;

12) The entry system will provide conditions compatible with BVS deployment requirements;

13) The external shape of the entry system capsule will be a large angle blunted cone;

14) Sterilization requirements as defined in JPL Specifications Nos. XSO-30275-TST-AJ and GMO 50198ETS-A shall be used.

The above technical guidelines permitted the buoyant station concept to be considered within the framework mission of specific modes and opportunities. The value of such an approach -- aside from extending the understanding of the buoyant station feasibility and further delineating the problem areas -- was to assess which mission modes best complement, or are complemented by, the buoyant station.

Several of the listed guidelines were modified early in the study -- primarily on the basis of the better understanding of Venus that resulted from the Venus Mariner mission in 1967 and the Russian mission in the same year. The guideline modifications are as follows:

1) The concept of using the minimum weight BVS developed under Contract NAS1-6607, was discarded on the grounds that the 20-lb science complement was no longer reasonable in the light of the Russian and Mariner successes, which made a larger payload more appropriate for consideration. The baseline science payload was increased to approximately 58 lb.

2) The SLV 3C Centaur was deleted from further consideration early in the study because it severely limited the size of the BVS under consideration -- particularly for the orbiter missions.

3) The atmospheric models of NASA SP-3016 were not used because better data were available and could be applied with reasonable assurance of its validity (see appendix A).

Finally, although not specified in the guidelines, it was decided to float the buoyant station within, rather than above the clouds because the scientific value of the mission would be significantly increased. Also a dual-altitude mission performed initially above the clouds was considered with a second phase of descent and floatation at lower altitudes (see appendix E).

At the end of the second month of the study, the recommendation was made to identify the entry-from-orbit concept and the flyby mission with direct-to-earth-communication as the approved configurations for the orbital and flyby modes respectively. The remaining design effort was concentrated on these configurations. The Venus Mercury mission was added in the seventh month as was the dual-altitude concept.

Volume I of this report, Mission Summary Definition and Comparison, summarizes the approved mission configurations so the mission modes and opportunities under consideration can be compared. Volume I includes a Preliminary Plan (Cost and Schedule) for each mission and an identification of mandatory or desirable technology effort related to the BVS concept.

Volume II, Trajectory Analysis for 1972 and 1973 Mission, presents the mission analysis performed for the three missions. A baseline is defined for each mission, including interplanetary trajectories, orbital parameters and flyby, approach, and atmospheric entry trajectories.

This volume, Configuration Definition, defines and documents the approved configuration for each mission. Included are the buoyant station entry system and modifications to the spacecraft.

The authors of each appendix to this volume are as follows:

Appendix A - Allan R. Barger;

Appendix B - H. Edward Sparhawk and John R. Mellin;

Appendix C - Ronald E. Frank and Jack D. Pettus;

Appendix D - **Allan R. Barger, Jack D. Pettus and Paul Reznicek;**

Appendix E - Patrick C. Carroll and Robert W. Stoffel;

Appendix F - James E. Cole and Ted H. Tucker;

Appendix G - James W. Berry;

Appendix H - Walter F. Hane and Jack D. Pettus.

SYMBOLS

accel	accelerometer
AGC	automatic gain control
APC	automatic phase control
A/S	aeroshell
Aux	auxiliary
B_E	entry ballistic coefficient
Bio	biological
bps	bits per second
BVS	Buoyant Venus Station
BW	bandwidth, Hz
Chrom.	chromatograph
DHS	data handling system
DSIF	Deep Space Instrumentation Facility
DSN	Deep Space Net
DSS	data subsystem sequencer
E	entry
EPD	Engineering Planning Document
F	frequency, Hz
F_s	square wave at frequency F
f_s	sine wave at frequency f
FSK	frequency shift key
MFSK	multiple frequency shift key

NASA	National Aeronautics and Space Administration
N/B	noise power density, watts/cycle
NRZ	nonreturn to zero
PCM	pulse code modulation
P_e^b	bit error probability
PN	pseudorandom noise
Press.	pressure
PSK/PM	phase shift key modulated subcarrier phase modulating a carrier
$P_{WE\ max}$	probability of word error
RTG	radioisotope thermoelectric generator
S	separation
S/C	spacecraft
SCR	silicon controlled rectifiers
Seq	sequencer
S/N	signal-to-noise ratio
$SN/_{in}$	signal-to-noise ratio in
$SN/_o$	signal-to-noise ratio out
Spec	spectrometer
SPS	samples per second
St	staging
Sync.	synchronization
T	temperature, °K
Temp.	temperature

Tr Sw	transfer switch
T_{sys}	system temperature, °K
TV	television
TW	time bandwidth product
VSWR	voltage standing wave ratio
Xducer	transducer
Xmtr	transmitter
α_B	spacecraft relay antenna aspect angle
α_{B1}	spacecraft antenna boresight angle referenced to line of apsides, degrees
α_{B2}	sum of α_{B1} and α_B
α_P	BVS antenna aspect angle BVS to S/C
β	targeting angle, central angle measured from entry point to periapsis, degree
ΔV	velocity increment
ϵ	polarization loss
γ_E	entry flight path angle, negative down, deg
λ	wave length (communications geometry)
λ	lead angle, central angle between the spacecraft and the entry vehicle at entry, deg (flight mechanics)
φ	for antenna gain patterns-angle normal to plane of widest beamwidth, degrees
ρ_c	communication distance, A.U.
θ	modulation index, radians
$2B_{LO}$	effective tracking loop bandwidth at threshold, H_z

BVS/ENTRY SYSTEM

GENERAL DESCRIPTION

A BVS/entry system has been defined in detail for three specific missions (illustrated in fig. 1): 1972 flyby mission, 1973 orbital mission, and 1973 Venus/Mercury mission.

All missions after entry are generally similar and follow the sequence of separation, deployment, and inflation shown in figure 2. The BVS/entry system consists of the following elements as shown in figure 3:

1) Buoyant station (BVS);
2) Subsonic probe;
3) Aeroshell and equipment attached to aeroshell;
4) Propulsion module;
5) Biocanister;
6) S/C adapter for attachment of capsule.

The BVS, generally similar for all missions, supports a 175-lb gondola under hydrogen gas-inflated, super-pressure balloon floating 58 km above the surface. The gondola contains a 58-lb complement of experiments. Communications are either relayed to a companion orbiter spacecraft or direct to earth. The flyby missions in 1972 and 1973 use the direct link during the floatation phase, while the orbital mission uses the relay mode.

The aeroshell houses the BVS, subsonic probe, and equipment mounted to the structure that is not required after BVS separation from the aeroshell. An 8.5-ft-diam blunted cone shape is used for the orbital and 1972 flyby missions with an elastomeric heat shield. The Venus/Mercury mission results in a higher entry velocity, 43 500 fps, requiring a high density (carbon phenolic) heat shield material. The aeroshell for this mission was reduced to a 7.0 ft diam. Because of the base heating associated with atmospheric entry of Venus, and afterbody, or cover, is required, which has a heat shield covering.

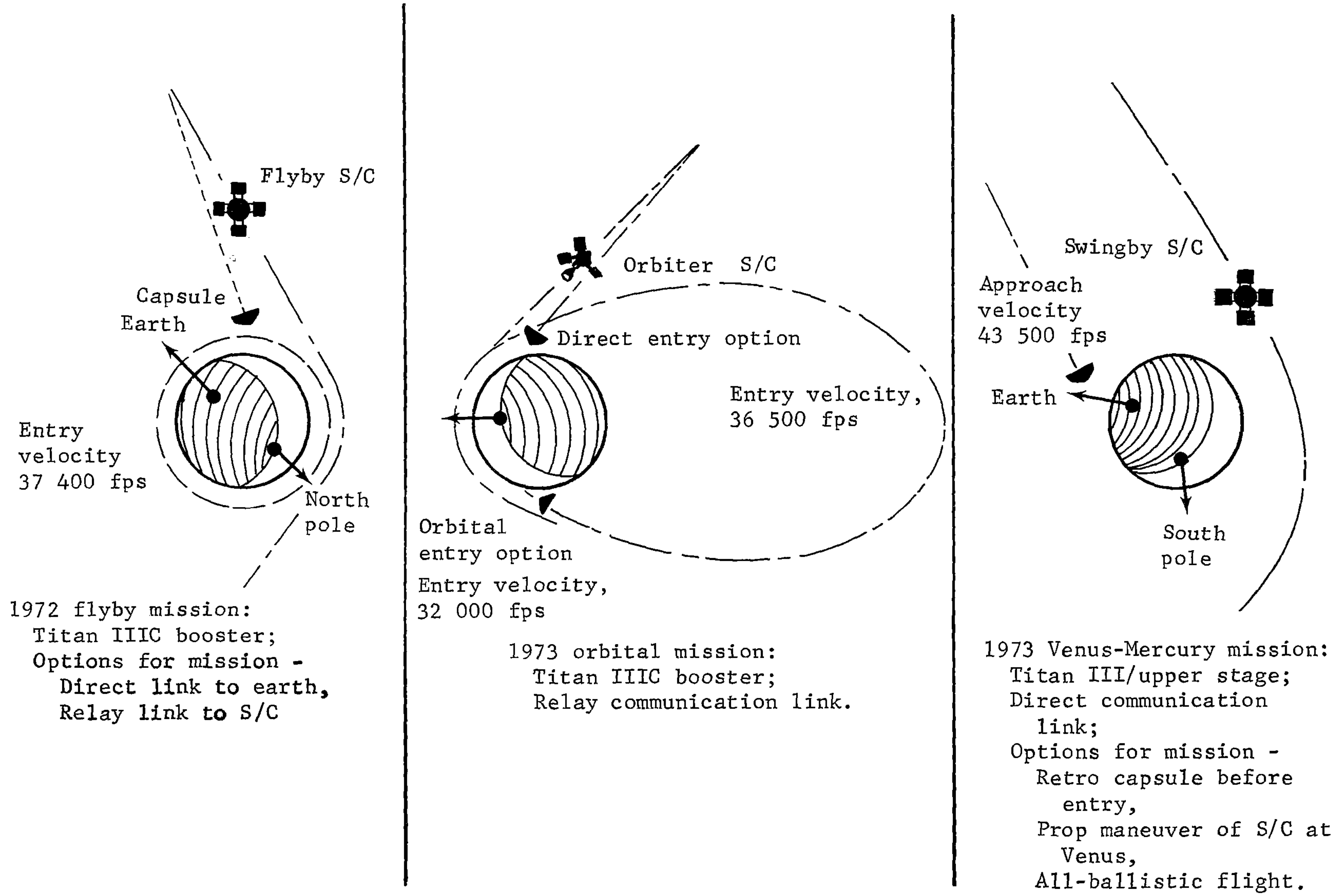


Figure 1.- BVS Missions

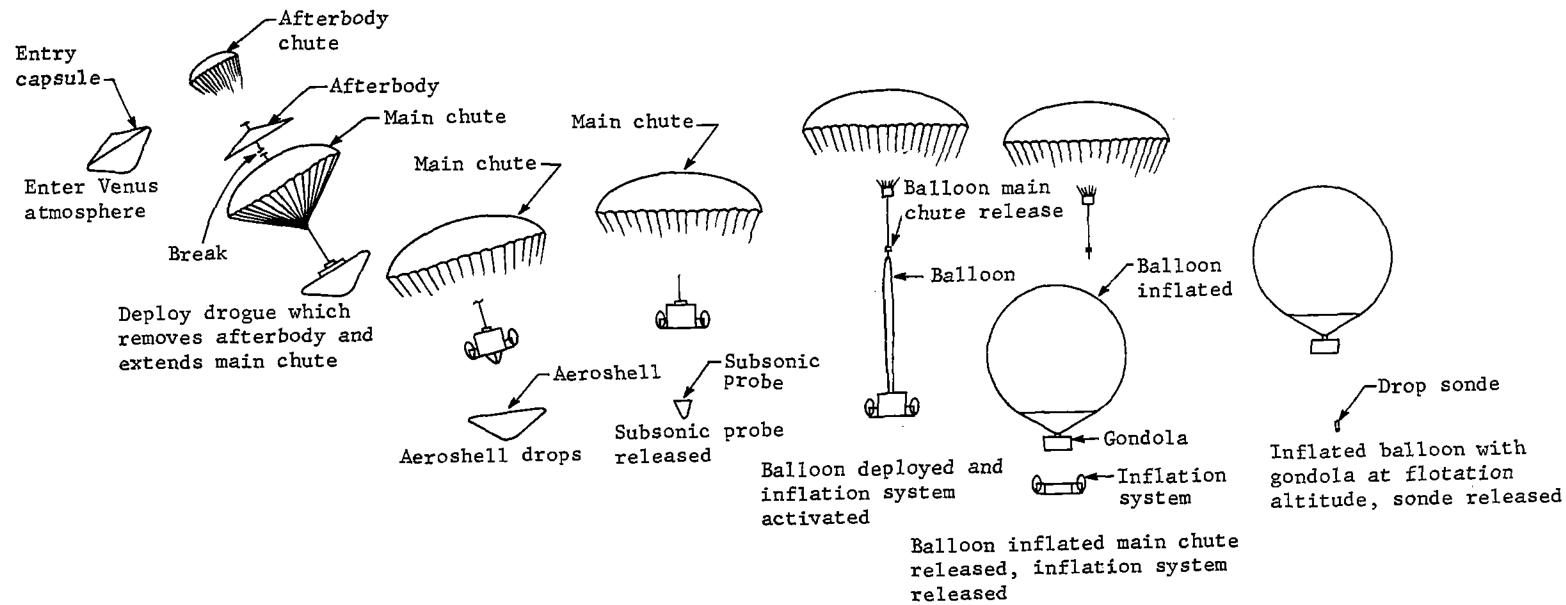
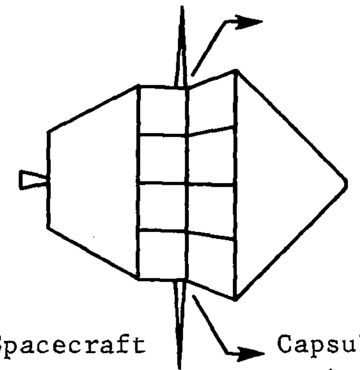
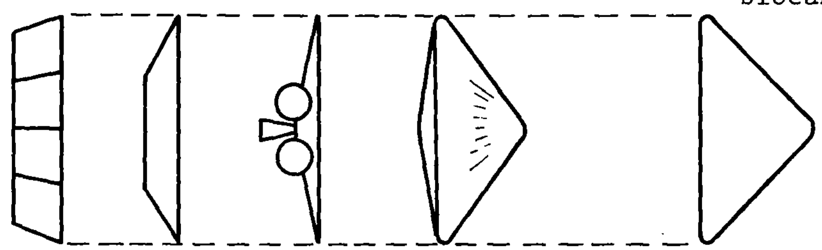


Figure 2.- BVS Deployment Sequence



Planetary vehicle = spacecraft and capsule system.



Capsule system = adapter (S/C to capsule), biological canister (aft), capsule propulsion module, capsule (BVS/entry vehicle), and forward biological canister.

<u>Note</u>: BVS/entry vehicle is the preverred nomenclature, but capsule can be used as equivalent.

Figure 3.- Nomenclature

A propulsion module is required to deflect or deorbit the capsule after separation from the S/C, supplying 50 mps, 93 mps, and 250 mps to the capsule for the 1972 flyby, 1973 Venus/Mercury swingby, and 1973 orbital missions, respectively. This module is separated immediately after firing, before atmospheric entry of the capsule.

The biocanister is a biological barrier that ensures the sterility of the capsule following heat sterilization. The canister consists of a forward and aft section. The forward section is pyrotechnically separated from the aft canister just before capsule-S/C separation. The aft canister is integral with the capsule-S/C adapter.

Drop Sonde

Two small drop sondes are dropped from the buoyant station to the surface after the station has been deployed. Pressure temperature and water vapor are measured. These data, acquired by each sonde as it descends to the surface, are relayed to the buoyant station via a frequency shift key telemetry link operating at 1500 MHz. The antennas are a helix located on the bottom ot the station gondola and a curved cross-dipole located on the top of the drop sonde. The sonde data system operates at 1 bps. The BVS, in addition to the 1500 MHz antenna, requires a 1500 MHz FSK receiver, a bit synchronizer, and core storage of at least 1800 bits.

Subsonic Probe

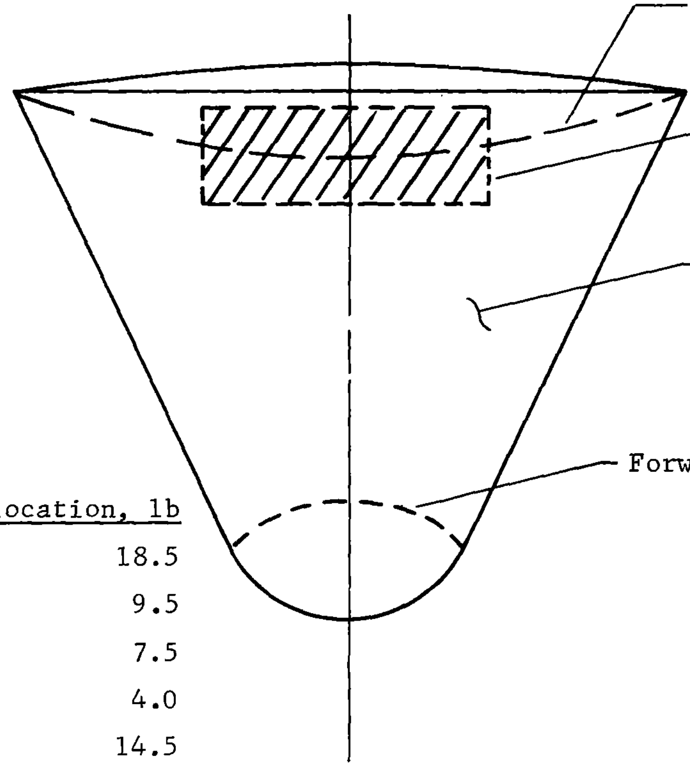
The subsonic probe has an independent mission of sampling the lower and near-surface atmosphere during its ballistic flight to the surface. This probe is an attractive addition to the basic BVS mission, but has been treated as an independent experiment and complete subsystem details have not been defined. The probe has been given an 85 lb weight allocation, of which 18.5 lb are allocated to science.

The probe is shown in figure 4 with its conical shape and ballistic coefficient of 6.0 slugs/ft^2. This produces a nominal flight time of 30 minutes to the surface based on release from the BVS inflation module at an altitude of approximately 70 km.

85-lb weight allocation

B = 6.0 slugs/ft^2

Flight time of 30 minutes to surface



Element weight, allocation, lb	
Science	18.5
Radio	9.5
Data management	7.5
Power (battery)	4.0
Thermal control	14.5
Structure	31.0
Total	85.0

Note: Thermal control is accomplished by phase change material plus radiation shields and insulation.

Figure 4.- Subsonic Probe

The construction concept is based on an evacuated equipment compartment that requires pressure bulkheads both forward and aft. The cross slot, cavity-backed antenna is protected by a radome.

Thermal control is accomplished by a combination of phase change material such as tetracosane, a parafin with a melting temperature of 124°F and a heat of fusion of 109 Btu/lb, in conjuction with radiation shields and fiberous insulation. This subsystem is estimated to weigh 14.5 lb.

The probe communication line is directly to the spacecraft for the orbital and flyby missions. The relay link is accomplished with a uhf, 400 MHz transmitter operating at 10 W rf, sending 240 bps of scientific data. There is one data format for the entire mission. The Venus/Mercury mission requires an S-band link directly to earth. A 20 W transmitter sends 120 bps to the DSN uncoded.

A complement of experiments is shown in table 1 for this type probe on the light side of the planet. The remaining subsystem weight allocations are shown in figure 4.

Deep Space Net Support

For the BVS orbital mission there are no direct communications interfaces with the BVS since all data is relayed through the S/C. Therefore support for the BVS would be equivalent to supporting a science experiment onboard the S/C and the station configuration would be equivalent to that for the Venus orbiter spacecraft.

For the flyby missions, however, there is a direct communications interface with the BVS and some discussion of the Earth-based subsystems is necessary.

TABLE 1.- SCIENCE COMPLEMENT FOR SUBSONIC PROBE

Measurement	Quantity	Weight allocation, lb	Power allocation, W
Static pressure	2	1.0	2.0
Dynamic pressure	1	0.7	1.0
Temperature	4	2.5	2.0
Density (β-source and detector)	1	0.9	0.5
Light backscatter (sources and detectors)	1	1.5	4.0
Visual light (3 channels)	1	2.0	1.5
IR light (3 channels)	1	1.6	3.0
Total insolation	1	1.0	1.0
Mass spectrometer	1	7.0	8.0
Probe Total		18.2	23.0

Venus flyby DSN configuration.- The direct communications interface with the BVS for the Venus flyby mission occurs immediately following the deployment of the buoyant station. Both a direct S-band command and a telemetry link are used. Each conforms to the standard station configuration in that the command link uses the standard Mariner 1969 configuration and the telemetry is standard for the Multiple Mission Telemetry Subsystem (MMTS) (refs. 2 and 3) presently being developed for the network by JPL.

For the command link an 85-ft antenna and a 100 W transmitter are required. For the telemetry link a 210-ft antenna is required.

At the time of entry through deployment of the BVS plus 20 minutes, both the flyby S/C and the BVS require direct two-way communications with a DSN station. Therefore, one station must support the S/C and another station must support the BVS during this period. This means that encounter must occur while two Earth stations are in view of the planet.

Venus/Mercury flyby DSN configuration.- The command and telemetry support configuration for the BVS for this mission is the same as for the Venus/flyby mission except that direct communications coverage in near real time plus predetection recording of the telemetry is required during entry. A 48 Hz receiver carrier tracking loop bandwidth is to be used instead of 12 Hz as for the Venus flyby. Simultaneous two station coverage would be required for this mission also.

ORBITER MISSION

The statement of work specified a mission mode to be considered with an orbiter spacecraft using a capsule (BVS/entry vehicle) entry-from-orbit mode, or a capsule entry-from-approach mode before orbital insertion of the spacecraft. These two options were considered in an early phase of the study. At this time the entry-from-orbit mode was recommended and approved for further definition and documentation, primarily because of the entry environment encountered with this mission. This permitted consideration of a wide range of entry environments from approximately 32 000 fps for this mission to 43 500 for the Venus/Mercury opportunity. It was recognized that many other facets of the entry-from-approach mode would ultimately be considered when the flyby mode was studied. The mission modes are compared in table 2. The guidelines under which this mission is defined are given in table 3.

TABLE 2.- ORBITAL MISSION MODE COMPARISON

Mission parameter	Orbiter-mode (1973) Entry from orbit		Entry from approach
Launch period	11/1	thru	11/21
Arrival dates	4/13	thru	4/19
Trajectory type	II		II
Max C_3, $(km/sec)^2$	8.36		8.36
Max V_{HE}, km/sec	4.32		4.32
Booster	Titan-IIIC		Titan-IIIC
Weight to planet, lb	3700		3700
Mission margin	125		630
Eccentricity	.80		.80
Periapsis altitude, km	2000		2000
Orbital period, hr	25		25
Spacecraft weight, lb	[a]940		[a]1155
Entry velocity, fps	32 000		36 500
Allowable capsule weight at entry, lb	840		[b]1010
Aeroshell, etc.	440		510
Allowable for BVS	400		500

[a]Weight in orbit.

[b]Limited by ballistic coefficient of 0.40 with 8.5 ft diam, blunt cone aeroshell.

TABLE 3.- ORBITAL MISSION GUIDELINES

Mission mode	Orbit of Venus
Launch opportunity	1973
Launch vehicle	Titan IIIC or Atlas/Centaur
Spacecraft	Modified lunar orbitor as defined by NASA CR-66302
Buoyant station	Modified minimum weight BVS investigated under Contract NAS1-6607
External entry shape	Large angle blunt cone
Communications	Relay via S/C. Assume a full complement of Deep Space Net Stations are available for support of mission.

The major differences between the entry-from-orbit mission and the entry-from-approach mission are the less severe entry environment, the more severe weight restriction on the BVS/entry vehicle, and available targeting and associated anticipated wind drift patterns in the floatation mode for the entry-from-orbit mode. A major consideration and restrictive requirement for the orbital missions is the selection of an orbit with sufficient stability to meet the 50-year quarantine requirement discussed in detail in Volume II.

Mission Description

The baseline mission shown in figure 5 has a type II transfer trajectory lasting approximately 165 days with a launch period of 1 November to 21 November 1973. At Venus encounter, the planetary vehicle is placed in a 25-hr orbit (see table 4). The vehicle performs its scientific mission during several orbits, after which time on command from Earth, the entry vehicle is separated and enters on the light side of the planet.

The entry site was selected to be compatible with the constraints of -30° entry angle, 30-minute view time with orbiter after entry and 250 mps deorbit impulse. The spacecraft 50-year orbit lifetime was an additional constraint limiting this mission. Figure 6 shows the entry site constraints and defines the location of five representative entry points. Figure 7 shows the effect of wind velocities where the drift time of the BVS to the terminator lies between 61 and 350 hr. Note that the BVS drifts generally parallel and out of the orbiter plane, which minimizes the problem of determining the position of the BVS from the S/C.

The overall mission weight summary is given in table 5, which indicates the 352-lb mission margin for the Titan IIIC launch vehicle. Additional detail on this mission is given in Volume II of this report.

Mission Sequence

The overall mission sequence is shown in figure 8 and the pertinent points are discussed below.

Prelaunch, launch, and cruise.- Before liftoff, power is applied to the BVS and subsonic probe telemetry subsystems and the subsystems are monitored. The capsule system is in a passive state except for periodic status monitoring, continuous battery charging, and heaters for thermal control, all controlled by the spacecraft, until six minutes before separation from the spacecraft.

Orbit insertion.- The planetary vehicle is oriented for proper orbit insertion, the velocity increment provided by the spacecraft propulsion module, and the orbit attained are shown in figure 9. The spacecraft then performs its orbital scientific mission from the orbit, which satisfies the 50-year quarantine requirement.

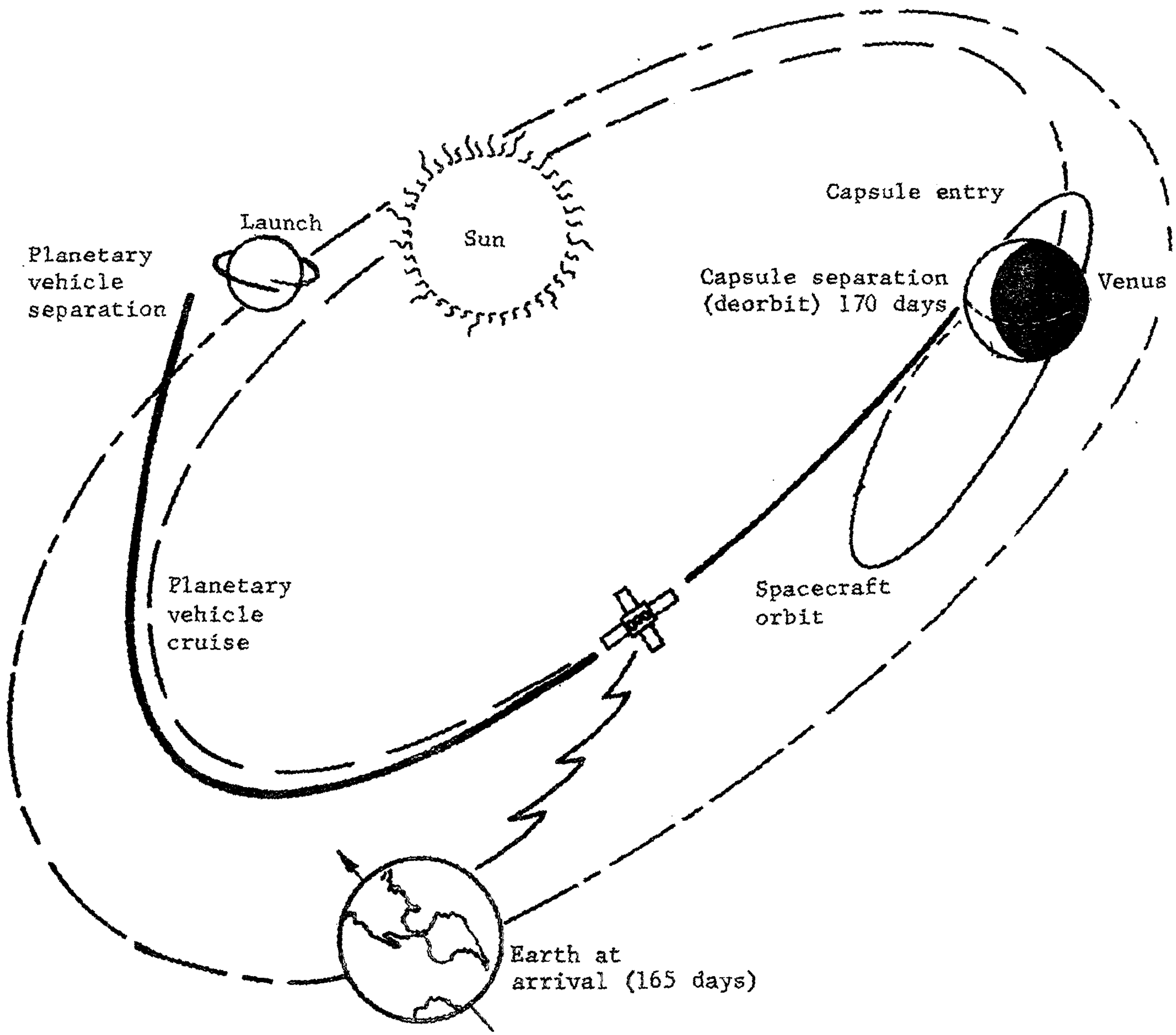
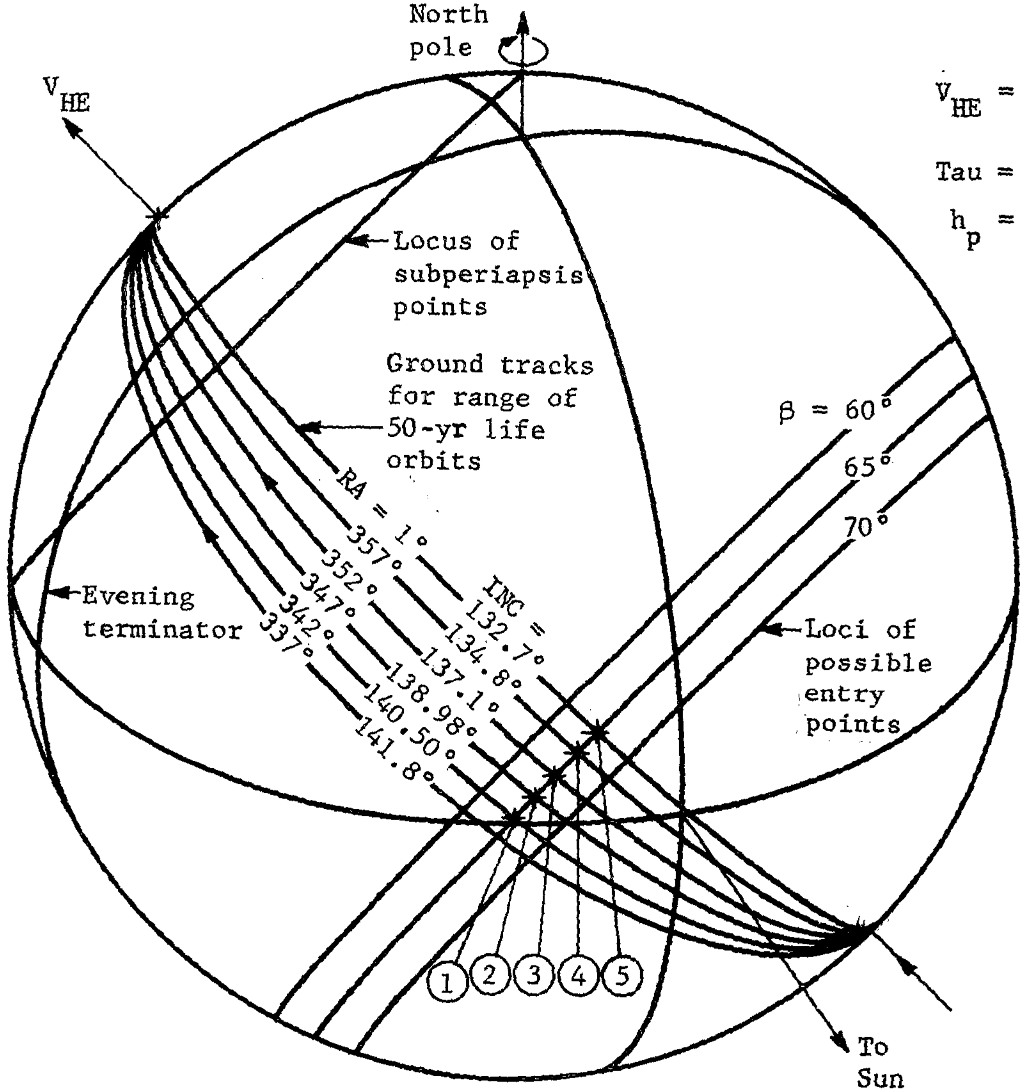


Figure 5.- 1973 Mission Profile for BVS

TABLE 4.- 1973 BVS MISSION SUMMARY

Orbiting spacecraft (modified Lunar orbiter)
661 lb useful weight in orbit
$\epsilon = 0.8$
$P = 25$ hr
$H_P = 2000$ km
Permits Earth occultation
Relay link for BVS and subsonic probe
Entry vehicle
$V_E = 32\ 000$ fps
$\gamma_E = -30°$
$B_E = 0.33$ slugs/ft^2
10 hr coast
Deorbit impulse = 250 mps
400 lb BVS
In-the-cloud mission (R = 6108 km)
$P_A = 612$ mb (~9 psia)
$T_A = 70°F$
Deployed approximately 15° from subsolar
58 lb of science (175 lb gondola)
Minibio laboratory
ATM measurements
Radar altimeter
Relay link to S/C
Hydrogen inflated superpressure balloon
Four manifolded tanks
Blowdown inflation
Supported by parachute during inflation



V_{HE} = 4.32 km/sec (at 35.6° N lat, 222.4 long)

Tau = 46.86°

h_p = 2000 km

Entry Points

	Latitude	Longitude
1	1.1°	340.5°
2	3.6°	342.7°
3	6.3°	345.1°
4	9.1°	347.7°
5	11.4°	350.2°

Figure 6.- 1973 Type II Orbiter Mission Encounter Geometry for First Day of Launch Window

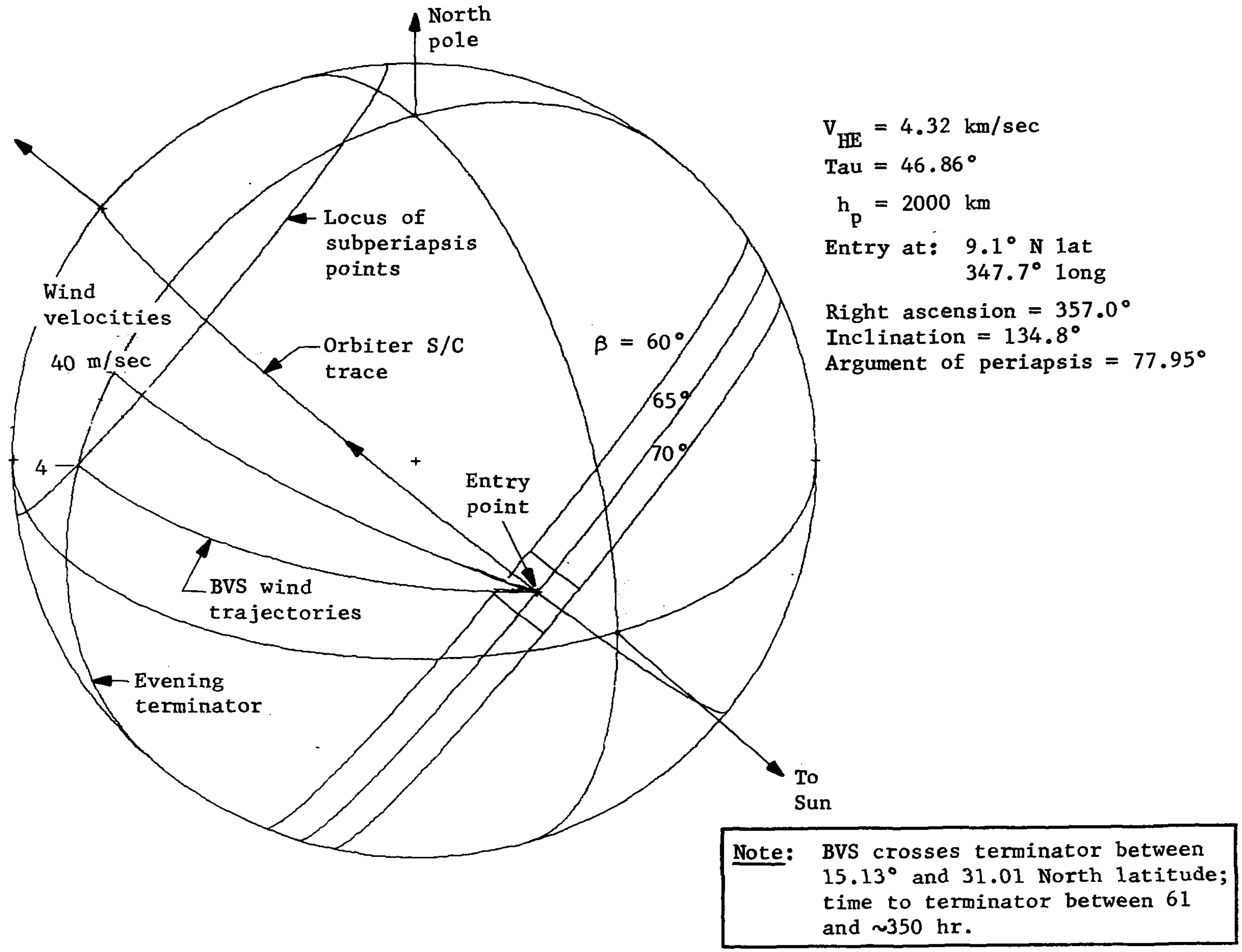
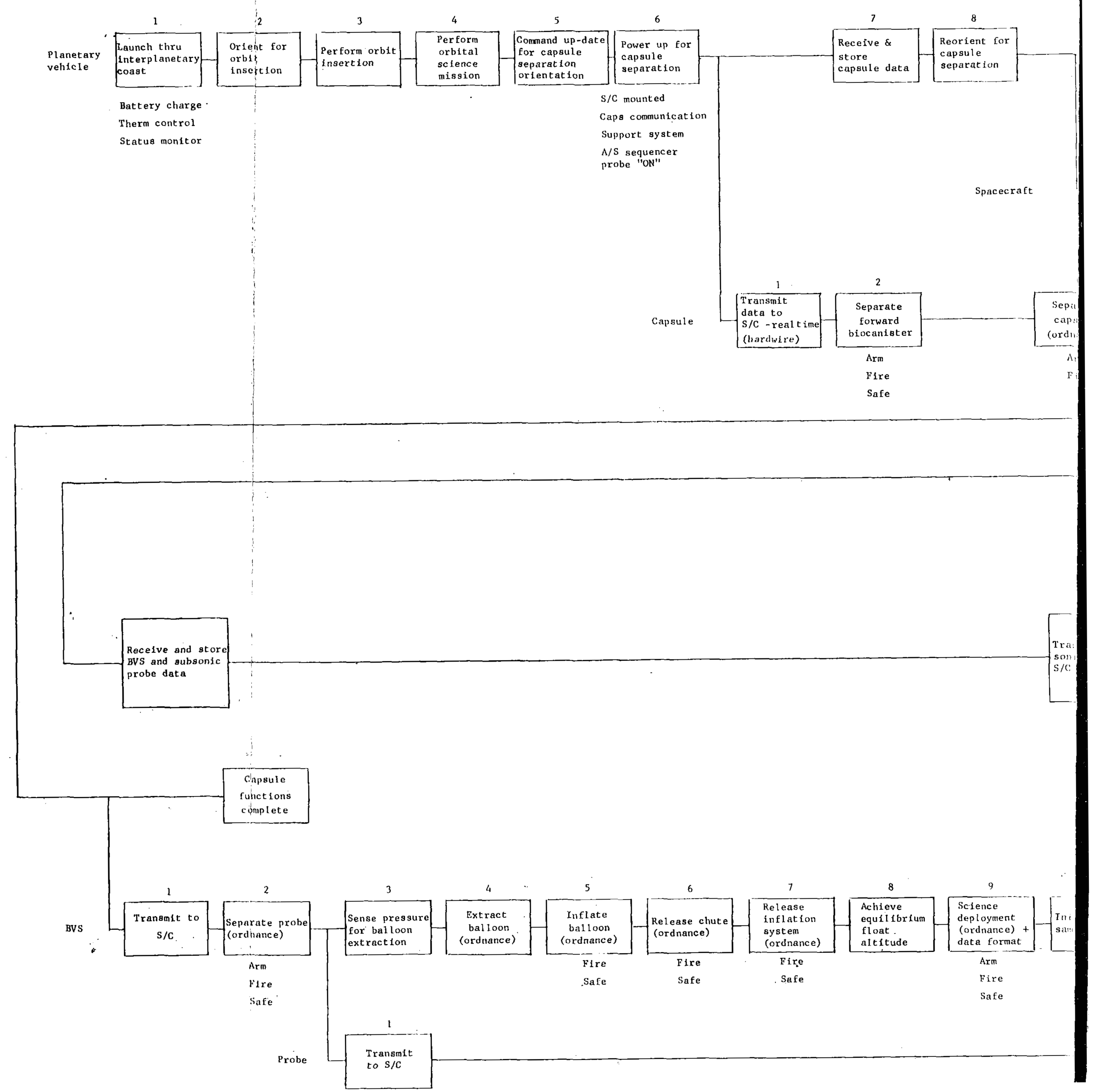


Figure 7.- BVS Wind Drift Trajectories for Nominal Entry Point, 1973 Orbiter Mission

TABLE 5.- BVS ORBITAL MISSION WEIGHT STATEMENT

Launch vehicle, Titan IIIC (Article 19 performance)				
C_3 = 8.36 $(km/sec)^2$ Maximum over 20-day launch period				
Capability				3775 lb
Planetary vehicle at cruise				(3423)
Spacecraft			2298	
Useful weight in orbit		661		
BVS/entry vehicle system			1125	
Entry weight		827		
Mission margin				352
BVS/entry vehicle system				(1125)
Capsule/spacecraft adapter (including biocanister)			153	
Propulsion module			145	
BVS/entry vehicle at entry			(827)	
BVS		403		
Subsonic probe		85		
Aeroshell with equipment		339		
Aeroshell	261			


Planetary vehicle
1
Launch thru interplanetary coast
Battery charge
Therm control
Status monitor
2
Orient for orbit insertion
3
Perform orbit insertion
4
Perform orbital science mission
5
Command up-date for capsule separation orientation
6
Power up for capsule separation
S/C mounted
Caps communication
Support system
A/S sequencer probe "ON"
7
Receive & store capsule data
8
Reorient for capsule separation
Spacecraft
Capsule
1
Transmit data to S/C -realtime (hardwire)
2
Separate forward biocanister
Arm
Fire
Safe
Receive and store BVS and subsonic probe data
Capsule functions complete
BVS
1
Transmit to S/C
2
Separate probe (ordnance)
Arm
Fire
Safe
3
Sense pressure for balloon extraction
4
Extract balloon (ordnance)
5
Inflate balloon (ordnance)
Fire
Safe
6
Release chute (ordnance)
Fire
Safe
7
Release inflation system (ordnance)
Fire
Safe
8
Achieve equilibrium float altitude
9
Science deployment (ordnance) + data format
Arm
Fire
Safe
Probe
1
Transmit to S/C

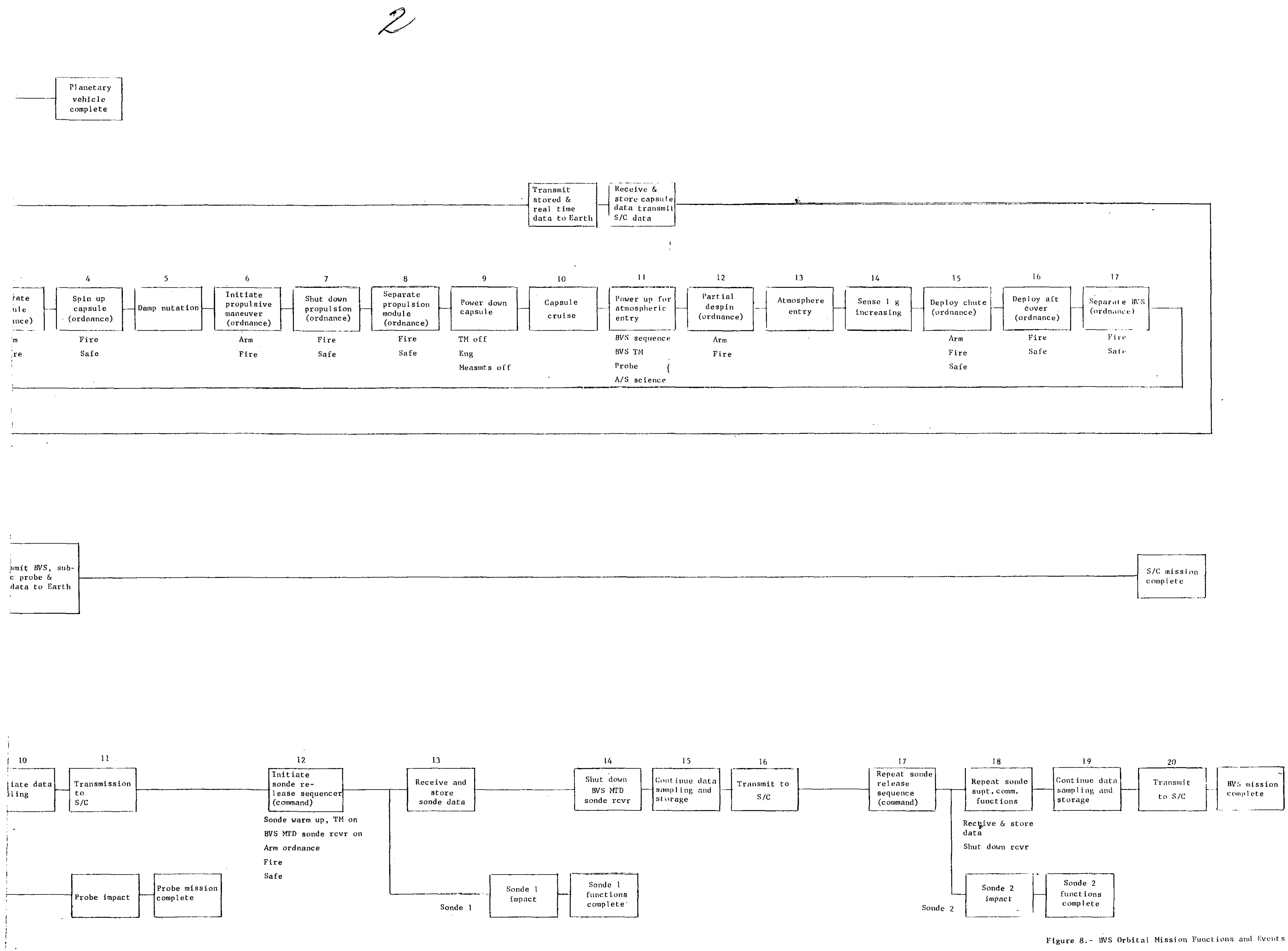


Figure 8.- BVS Orbital Mission Functions and Events

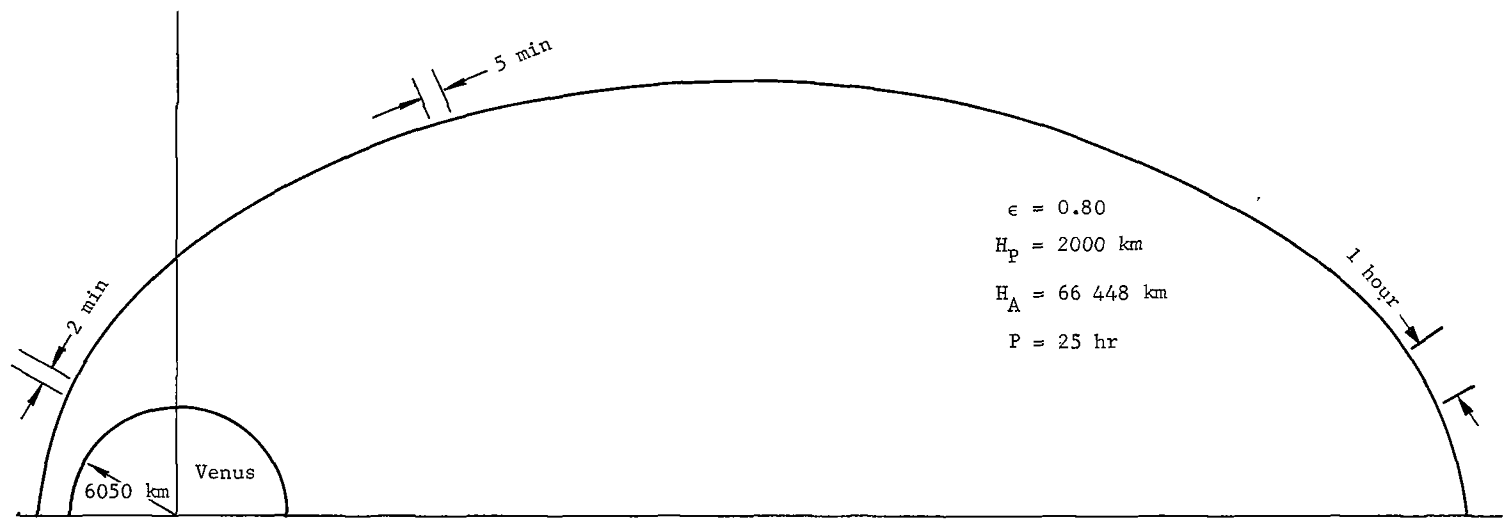


Figure 9.- Orbiter Spacecraft Orbit for BVS Mission

Capsule initiation.- On command from Earth to the S/C, the aeroshell equipment and BVS are powered up at separation minus 6 minutes. The capsule biocanister (forward section) is ejected at separation minus 4 minutes.

Spacecraft/capsule separation.- The spacecraft orients the capsule for ejection. The separation is realized by pyrotechnic operation powered from the capsule.

Capsule stabilization and deflection.- The capsule is spun up to 2.20 rad/sec followed by a 20-minute coast. The BVS is powered up at separation plus 19.5 minutes. A velocity increment of 250 mps maximum is then supplied, followed by propulsion module separation. The BVS transmits data to the S/C during this maneuver and shuts down at separation plus 25 minutes.

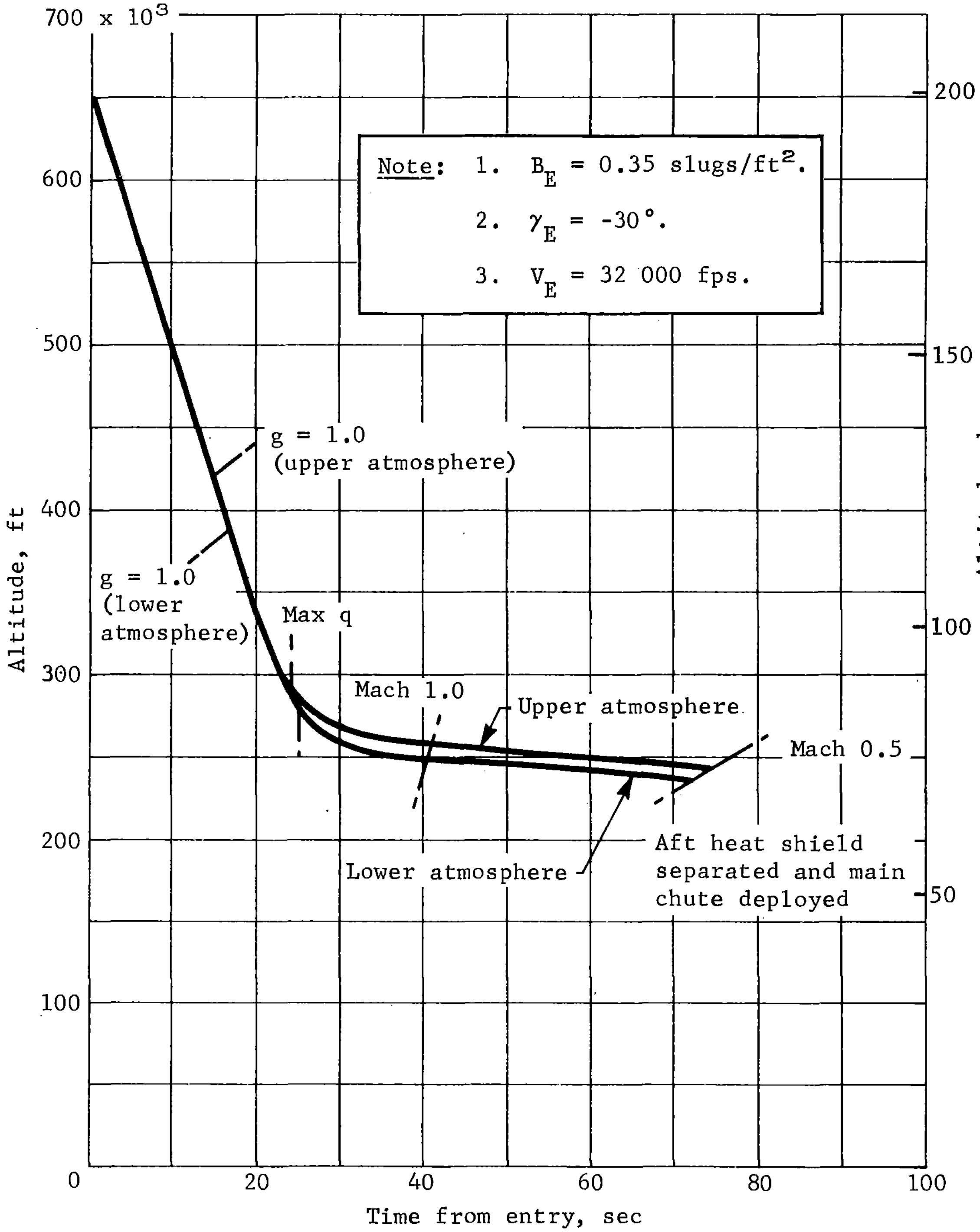
Capsule coast.- The capsule is powered down for the coast period (before entering the atmosphere). Five minutes before entry the BVS is powered up and the capsule partially despun to 0.5 rad/sec.

Atmospheric entry.- The BVS/entry vehicle enters the atmosphere and stores the entry data for 25 sec (during blackout) and plays these data out after blackout. The BVS deployment and subsonic probe ejection are based on a preselected time from 1.0 g. The entry and deployment environments are illustrated in figure 10.

BVS deployment/subsonic probe ejection.- Upon reaching Mach 0.50 (based on a preselected time from 1.0 g) the aeroshell afterbody is pyrotechnically separated using a parachute deployed by a mortar. This parachute deploys the BVS main parachute, which stages the BVS and subsonic probe from the aeroshell. The probe is powered up and released from the BVS and descends ballistically to the surface.

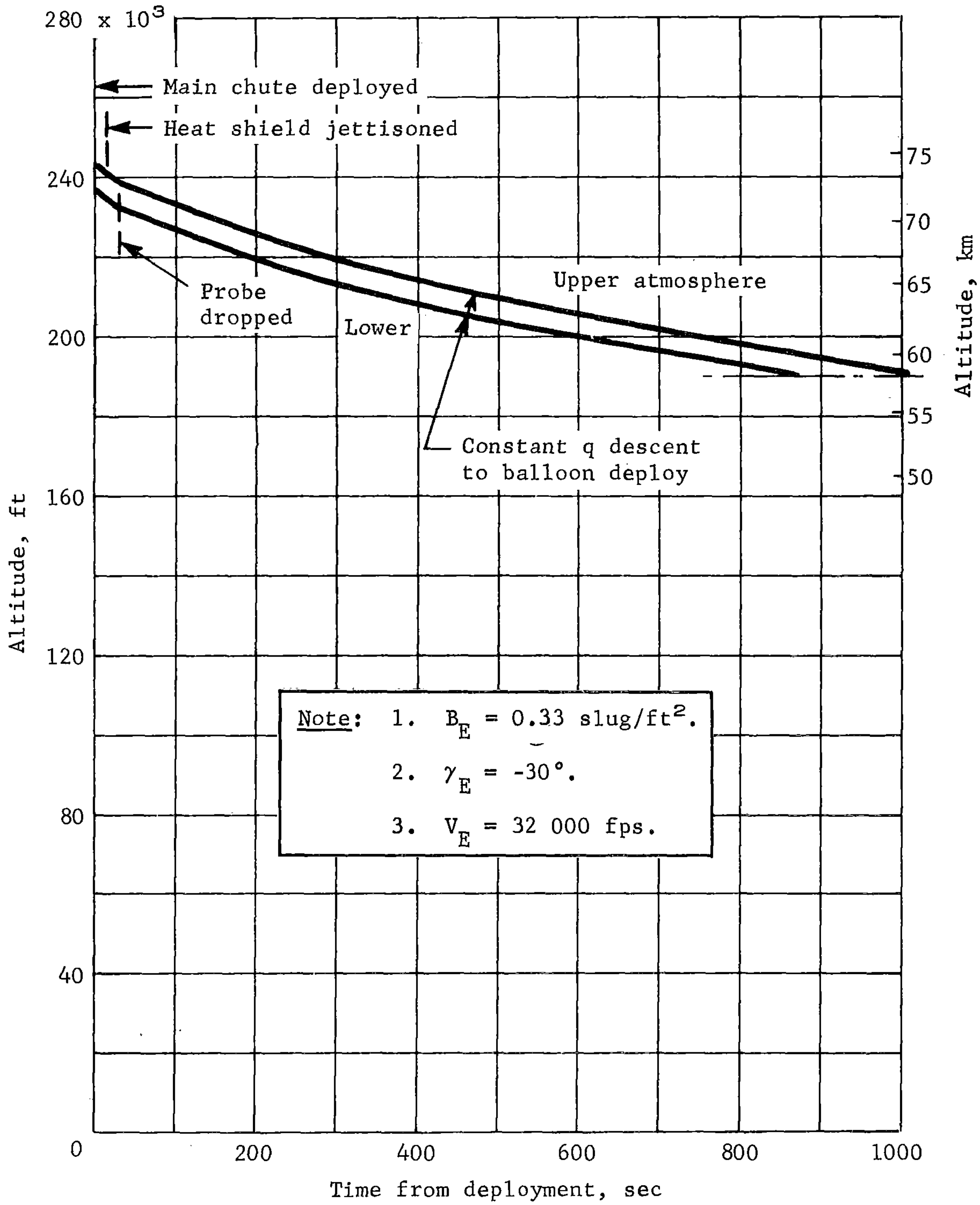
Upon sensing the proper atmospheric pressure, the balloon is extended from its canister and inflation is initiated. The high-pressure gas system blows down, inflating the balloon within 60 sec. The parachute is released followed by the release of the expended tankage and controls.

Throughout this deployment period the BVS relays data to the spacecraft. Also during this period and until impact, the subsonic probe relays its scientific data to the spacecraft.



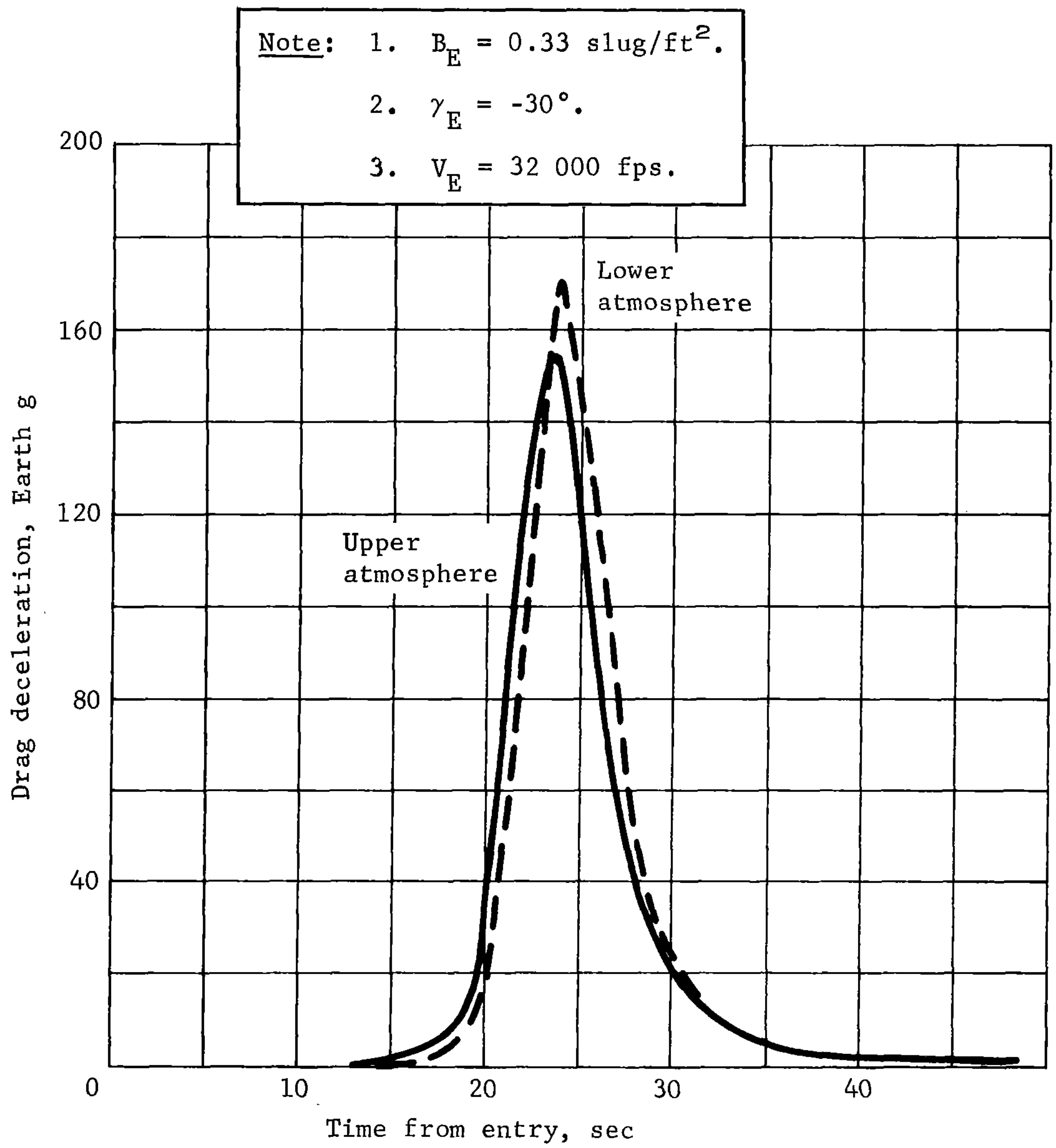
(a) Altitude vs Time from Entry

Figure 10.- BVS Reference Entry Trajectory



(b) Altitude vs Time from Deployment

Figure 10.- Continued



(c) Drag Decelerations vs Time from Entry

Figure 10.- Concluded

BVS flotation.- The BVS flotation mission is within the clouds (see table 6) drifting with the wind. Upon command via S/C a drop sonde is released to descend to the surface, relaying data to the BVS until impact. A second drop sonde is released on a second command. Each pass of the orbiter is sensed by the BVS and the data are relayed to the spacecraft for a nominal period of 10 minutes. A detail sequence of events for the mission is given in table 7.

TABLE 6.- BVS FLOTATION MISSION

Atmosphere, % of CO_2	Density, lb/ft^3	Pressure, mb	Temperature, °F	Radius from planet center, km
85	.0659	610.8	58.5	6107.5
90	.0659	612.4	70.2	6108.0
95	.0659	616.7	83.5	6108.4

TABLE 7.- BVS MISSION FUNCTIONS AND EVENTS

Function	Time	From system	To system
A. Planetary vehicle			
1. Launch through interplanetary coast			
a. Battery charge			
b. Thermal control			
c. Status monitor			
2. Orient for midcourse correction			
3. Perform midcourse correction			
4. Reorient for coast			
5. Orient for orbit insertion			
6. Perform orbit insertion			
7. Perform S/C orbital science mission			
8. Command update for capsule separation orientation			
9. Power up for capsule separation			
a. S/C MTD capsule communication support			
b. Aeroshell sequencer			
c. Aeroshell data encoder			
d. Aeroshell engineer measurements			
e. BVS			
10. S/C receive and store capsule data			
11. Separate forward biocanister	[a]S-00:04:00		
12. S/C reorient for capsule separation			
13. Capsule separation	S-00:00:00		
B. Spacecraft			
1. Reorient for orbital science mission			
2. Transmit stored and real-time data to Earth			
3. Receive and store capsule data, transmit S/C data			
4. Shut down S/C MTD capsule communication support system			
5. Transmit stored probe data			
C. Capsule			
1. Transmit data to S/C - real time (hardwire)	S-00:06:00	A/S sequencer	BVS data system sequencer (BVS DSS)
2. Separate forward biocanister (3 igniters for linear shaped charge)			
a. Arm	S-00:04:50	A/S sequencer	A/S pyro
b. Fire	S-00:04:00	A/S sequencer	A/S pyro
c. Safe	S-00:03:50	A/S sequencer	A/S pyro
3. Initiate data format A (BVS)	S-00:01:00	A/S sequencer	BVS DSS
4. Separate capsule (4 igniters)			
a. Arm	S-00:00:09	A/S sequencer	A/S pyro
b. Fire	S-00:00:00	A/S sequencer	A/S pyro
5. Spin up capsule (2 igniters)			
a. Fire	S+00:00:01	A/S sequencer	A/S pyro
b. Safe	S+00:00:03	A/S sequencer	A/S pyro
c. Power down BVS	S+00:05:00	A/S sequencer	BVS DSS
6. Damp nutation			
7. Power up BVS	S+00:19:30	A/S sequencer	BVS DSS
7a. Initiate propulsive maneuver (bi-propellant module)			
a. Arm	S+00:19:49	A/S sequencer	A/S pyro
b. Pressurize propellant (3 igniters)	S+00:19:50	A/S sequencer	A/S pyro
c. Fire (2 igniters and 2 solenoid valves)	S+00:20:00	A/S sequencer	A/S pyro
d. Safe	S+00:20:01	A/S sequencer	A/S pyro
8. Shut down propulsion			
a. Arm	S+00:24:16	A/S sequencer	A/S pyro
b. Fire (2 igniters and 2 solenoid valves)	S+00:24:17	A/S sequencer	A/S pyro
9. Separate propulsion module			
a. Fire (2 igniters)	S+00:24:30	A/S sequencer	A/S pyro
b. Safe	S+00:24:31	A/S sequencer	A/S pyro

TABLE 7.- BVS MISSION FUNCTIONS AND EVENTS - Continued

Function	Time	From system	To system
10. Power down capsule	S+00:25:00	A/S sequencer	BVS DSS
a. TM off			
b. Engine measurements off			
c. Only A/S sequencer operating			
11. Capsule cruise			
12. Power up for atmospheric entry	S+10:19:37		
a. BVS telecommunications (data format B)		A/S sequencer	BVS DSS
b. A/S science on		A/S sequencer	A/S power
13. Partial despin (1 igniter)			
a. Arm	S+10:19:38	A/S sequencer	A/S pyro
b. Fire	S+10:19:39	A/S sequencer	A/S pyro
c. Safe	S+10:19:40	A/S sequencer	A/S pyro
14. Atmospheric entry (1.0 g increasing)		BVS science	Sequencer
a. Initiate storage of data format C	S+10:24:37 [b](E-00:00:00)		
15. Deploy afterbody parachute (1 igniter) (M = 0.50)			
a. Arm	E+00:01:14	A/S sequencer	A/S pyro
b. Fire	E+00:01:15	A/S sequencer	A/S pyro
c. Safe	E+00:01:16	A/S sequencer	A/S pyro
16. Release afterbody[c] (3 igniters)			
a. Arm	E+00:01:30	A/S sequencer	A/S pyro
b. Fire	E+00:01:31	A/S sequencer	A/S pyro
c. Safe	E+00:01:32	A/S sequencer	A/S pyro
17. Release BVS and SS probe from A/S (3 igniters)			
a. Arm	E+00:01:46	A/S sequencer	A/S pyro
b. Fire	E+00:01:47	A/S sequencer	A/S pyro
(Capsule functions complete)			
D. BVS (from time of separation from A/S)			
1. Separate probe (SSP)(2 igniters)			
a. Arm BVS ordnance	[d]St+00:00:14	Sequencer	Pyro
b. Turn on SSP		Sequencer	SSP power
c. Fire	St+00:00:15	Sequencer	Pyro
d. Safe	St+00:00:16	Sequencer	Pyro
2. Switch to BVS data format C	St+00:01:00	BVS DSS	Telecomm
a. Format C is stored for 2 sec when BVS experiences greater than 1.0 g			
3. Switch to BVS data format D	St+00:06:00	BVS DSS	Telecomm
4. Sense ambient pressure	(Approx St+00:15:00)	Pressure switch	Sequencer
5. Extract balloon (1 igniter)			
a. Arm	[e]PP+00:00:01	Sequencer	Pyro
b. Fire	PP+00:00:02	Sequencer	Pyro
6. Inflate balloon (2 igniters)			
a. Fire (sock inflation)	PP+00:00:03	Sequencer	Pyro
b. Fire (balloon inflation)	PP+00:00:05	Sequencer	Pyro
c. Safe	PP+00:00:06	Sequencer	Pyro
7. Release parachute (1 igniter on parachute bridle)			
a. Fire	PP+00:00:40	Sequencer	Parachute pyro
8. Terminate inflation			
a. Arm	PP+00:00:57	Sequencer	Pyro
b. Fire	PP+00:00:58	Sequencer	Pyro
9. Release inflation system (3 igniters)			
a. Arm	PP+00:00:59	Sequencer	Pyro
b. Fire	PP+00:00:60	Sequencer	Pyro
c. Safe	PP+00:01:01	Sequencer	Pyro
10. Science deployment (1 igniter)			
a. Arm	PP+00:01:02	Sequencer	Pyro
b. Fire	PP+00:01:03	Sequencer	Pyro
c. Safe	PP+00:01:04	Sequencer	Pyro

TABLE 7.- BVS MISSION FUNCTIONS AND EVENTS - Concluded

Function	Time	From system	To system
11. Warm up science	PP+00:03:00	BVS DSS	Science
12. Store frame of format E	PP+00:03:00	BVS DSS	Telecomm
13. Store frame of format F	PP+00:03:10	BVS DSS	Telecomm
14. Transmit stored data	PP+00:03:30	BVS DSS	Telecomm
15. Turn off data system	PP+00:16:00	BVS DSS	Telecomm
16. Sample format E (repeat each hour)	PP+01:03:00	BVS DSS	Telecomm
17. Sample format F (repeat each 6 hours)	PP+06:03:00	BVS DSS	Telecomm
18. S/C beacon received	[f]R-00:00:00 (Approx S+25 hr)	BVS DSS	Telecomm
a. If in data collection cycle, complete cycle		BVS DSS	Telecomm
b. If not in data collection cycle, lock out cycle		BVS DSS	Telecomm
c. Data handling subsystem on, transmitter on, science on, doppler data unit on			
19. Format E transmitted in real time	R+00:03:10	BVS DSS	Telecomm
20. Format E stopped, transmit stored data	R+00:03:40	BVS DSS	Telecomm
21. Transmitter off, science off, switch to data format D	R+00:10:00	BVS DSS	Telecomm
22. Doppler data unit off	R+01:30:00	BVS DSS	Telecomm
23. Initiate sonde release sequence (command)	[g]SR-00:00:11	Spacecraft	BVS receiver
a. Sonde warm up, TM "ON"	SR-00:00:10	BVS DSS	Sonde
b. BVS MTD sonde data unit on	SR-00:00:09	BVS DSS	Telecomm
c. Arm (ordnance for release)	SR-00:00:01	BVS DSS	Pyro
d. Fire	SR-00:00:00	BVS DSS	Pyro
e. Safe	SR+00:00:01	BVS DSS	Pyro
24. Receive and store sonde data		BVS DSS	Telecomm
25. Shut down BVS MTD sonde receiver	SR+00:30:00	BVS DSS	Telecomm
26. Repeat steps 16 through 23			
27. Low power mode (command) (data format E)		Spacecraft or BVS power	BVS receiver or BVS DSS
28. Return to normal power mode (command)		BVS power	BVS DSS
E. Subsonic probe (from time of release from BVS)			
1. Science deployment (data format 1)		(Probe data to S/C)	
2. Probe impact (mission complete)	[h]SSPR+00:30:00		
F. Sonde 1			
Sonde 1 impact (mission complete)		(Sonde data to BVS)	
G. Sonde 2			
Sonde 2 impact (mission complete)		(Sonde data to BVS)	

[a] S stands for separation of capsule from spacecraft.

[b] E stands for atmospheric entry.

[c] This function extends the BVS parachute.

[d] St stands for staging of BVS from aeroshell.

[e] PP stands for proper pressure to extract balloon.

[f] R stands for spacecraft beacon received.

[g] SR stands for sonde release.

[h] SSPR stands for subsonic probe release from BVS.

General Configuration

The planetary vehicle shown in figure 11 in the cruise configuration consists of the spacecraft and entry capsule system. The spacecraft (orbiter) is based on a modified Lunar orbiter (ref. 1) and has 661 lb of useful weight in orbit including 109 lb of orbiter science. A larger thrust capability has been added (with an initial thrust-to-weight ratio of 0.65) to insert the planetary vehicle into orbit, thus effectively eliminating gravitational losses.

The BVS/entry vehicle along with the adapter, biocanister, and propulsion module make up the entry capsule system.

FLYBY MISSION

The statement of work specified consideration of a mission modes with a flyby spacecraft using a direct entry capsule with direct link and relay link via S/C communication modes between the buoyant station and Earth. These two options were compared early in the study and two relay mode concepts were considered. In the first concept the BVS/entry vehicle enters at the S/C anti-subperiapsis point on Venus and allows a relay link to be established with the S/C both before and after S/C planet encounter. The second concept has the BVS/entry vehicle entering on the far side of the planet so that BVS deployment occurs after the S/C has passed through periapsis and emerges from behind the planet. The BVS then communicates with the S/C as the S/C travels away from the planet on a course approximately parallel to the BVS vertical axis. Based on the factors summarized in table 8, the direct link mode was recommended and approved for further definition and documentation. This decision was based primarily on the fact that the direct link allows for a BVS mission duration of several days, while the relay links via S/C limit the BVS mission data return to the order of 15 to 20 hr. Certainly an attractive feature of the BVS is its potential long-duration mission, however, the relay link mode, via S/C, is feasible.

The guidelines under which this mission is defined are given in table 9.

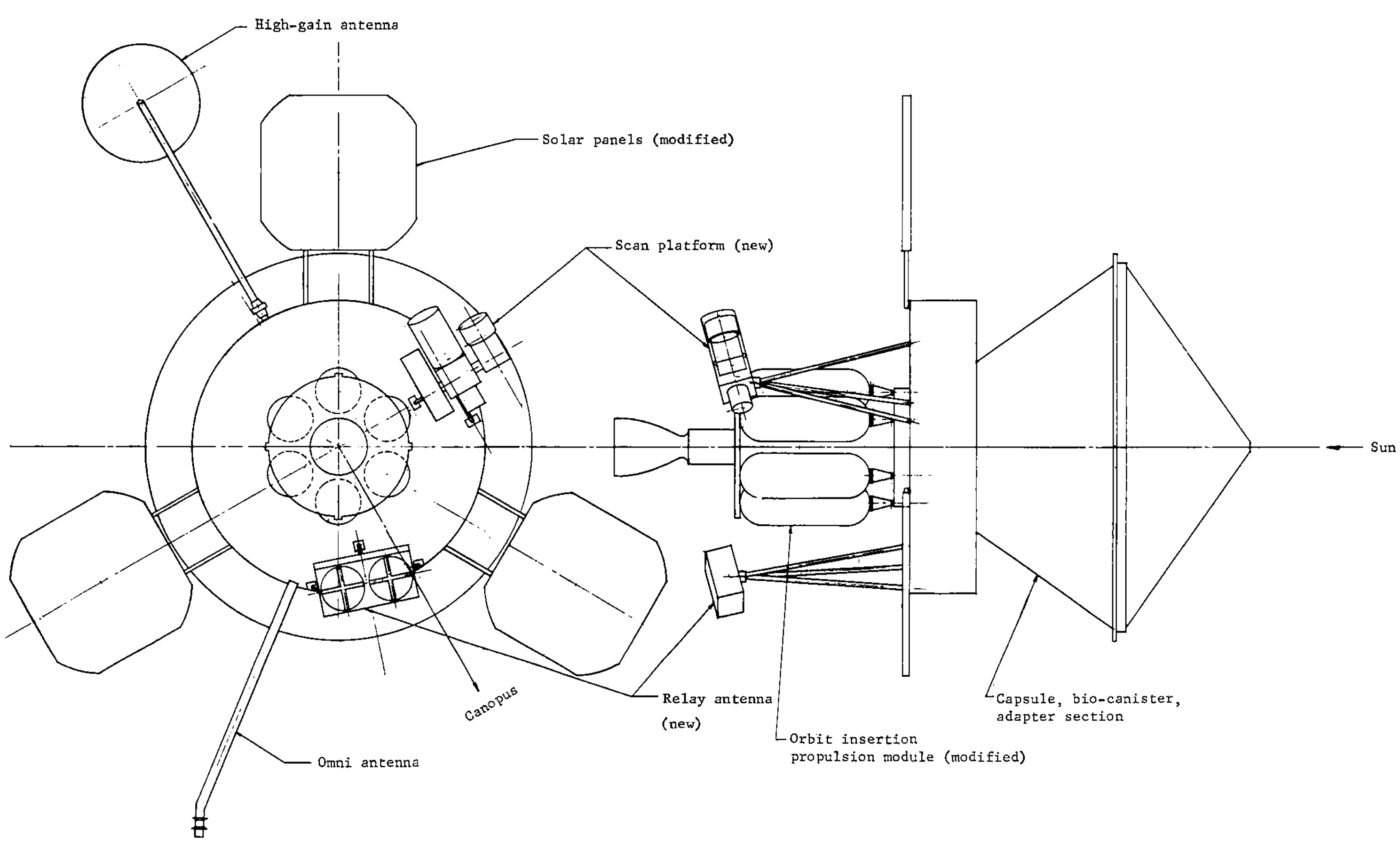


Figure 11.- Planetary Vehicle for Orbital Mission

TABLE 8.- BVS DIRECT VS RELAY COMMUNICATIONS COMPARISON

Factor	Direct Link		Relay link via S/C	
	Type I mission	Type II mission	Mission with pre- and post-encounter link	Mission with post-encounter link only
Communication time, hr	Limited by BVS drift and electrical energy		8	17
Communication range	0.5 AU	0.91 AU	153 000 km	340 000 km
Communication angles	20° or more above horizon		20° or less above horizon	Close to 90° above horizon
Transmitter power, W	20	30	40	40
Data rate, bps	30	10	10	10
Modulation	Coherent	Coherent	Coherent	Coherent
Entry considerations	Nominal entry parameters allow for good link		Severe entry angle required resulting in aeroheating problem	Nominal entry parameters but requires capsule reorientation (active ACS)
S/C operation	Not required	Not required	Must receive, store and relay BVS data	
S/C antenna	Not required	Not required	2-position conical helix	Fixed conical helix
DSIF interface	Must track BVS and S/C		Must track S/C only	Must track S/C only
Entry/deployment considerations	Short range (15 000 km), no acquisition search required, noncoherent modulation acceptable		Long-range (66 000 km), acquisition search with coherent modulation required	Long-range (54 000 km) acquisition search with coherent modulation required

TABLE 9.- FLYBY MISSION GUIDELINES

Mission mode	Flyby of Venus
Launch opportunity	1972
Launch vehicle	Titan IIIC or Atlas/Centaur
Spacecraft	Mariner 1969 as defined in Mariner Mars 1969 Functional Requirements
Buoyant station	Modified minimum weight BVS investigated under contract NAS1-6607
External entry shape	Large angle blunt cone
Communications	Relay via S/C vs direct to Earth. Assume a full complement of Deep Space Net Stations are available for support of mission.

The major points of difference between the direct and relay via S/C communication mode missions are (1) available targeting and associated anticipated wind drift patterns for the BVS; (2) communication period with BVS; (3) loss of communication with BVS when S/C travels behind planet; and (4) postencounter trajectory requirement for the S/C

The flyby mission does not present as severe a weight restriction on the BVS/entry vehicle or spacecraft as the orbital mission.

Mission Description

The baseline mission shown in figure 12 has a Type 1 transfer trajectory lasting approximately 123 days with a launch period of 1 April to 21 April 1972. Approximately 53 hr before planet encounter the BVS/entry vehicle is separated and deflected for entry at Venus. The S/C continues on a flyby of the planet and relays the entry and deployment data of the BVS and subsonic probe to earth after the S/C performs its scientific mission at Venus encounter (see table 10).

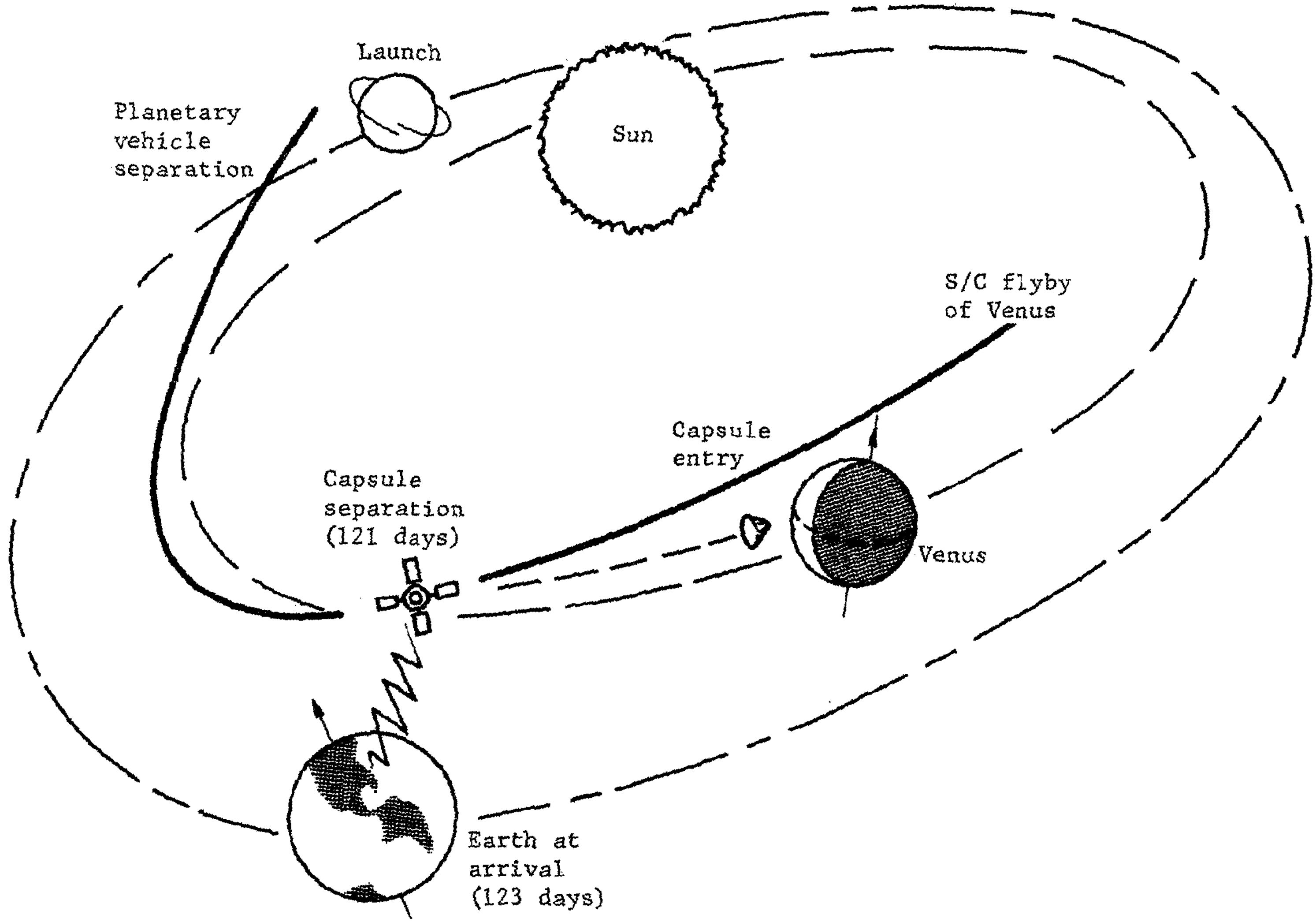


Figure 12.- 1972 Mission Profile for BVS

TABLE 10.- 1972 BVS MISSION SUMMARY

Flyby spacecraft (modified Mariner 1969)
882 lb at flyby
H_P = 2000 km
Relay link for BVS and subsonic probe (entry and deployment only)
Entry vehicle
V_E = 37 400 fps
$\gamma_E = -30°$
B_E = 0.33 slugs/ft^2
52.8-hr coast
Deflection impulse = 50 mps
Radius at Ejection = 1 000 000 km
400-lb BVS
In-the-cloud mission (R = 6108 km)
P_A = 612 mb (~ 9 psia)
T_A = 70°F
Deployed approximately 55° from subearth
58 lb of science (175-lb gondola)
Minibio laboratory
ATM measurements
Radar altimeter
Direct link to earth (30 bps)
Hydrogen inflated superpressure balloon
4 manifolded tanks
Blowdown inflation
Supported by parachute during inflation

The entry site was selected to be compatible with the constraints of -30° entry angle, not more than 60° from subearth for communications, and at least 20° from the terminator to allow an adequate mission before crossing the terminator with the balloon. Wind patterns were the main factor in selecting a point within the area defined by the above parameters. The higher wind velocities will result in a 17-hr drift to the terminator, while lower velocity will produce a time to the terminator of many hundreds of hours. The entry site is as near subearth as possible because loss of communications will end the mission abruptly, while the higher wind velocities will cause the BVS to cross the terminator independent of entry site within the target area. Figure 13 shows the entry and defines the location of three representative points. Figure 14 shows the effect of wind velocities within the model for the selected entry location.

The overall mission weight summary is given in table 11, which indicates the 300-lb margin for the Titan IIIC launch vehicle. Additional detail on this mission is given in Volume II of this report.

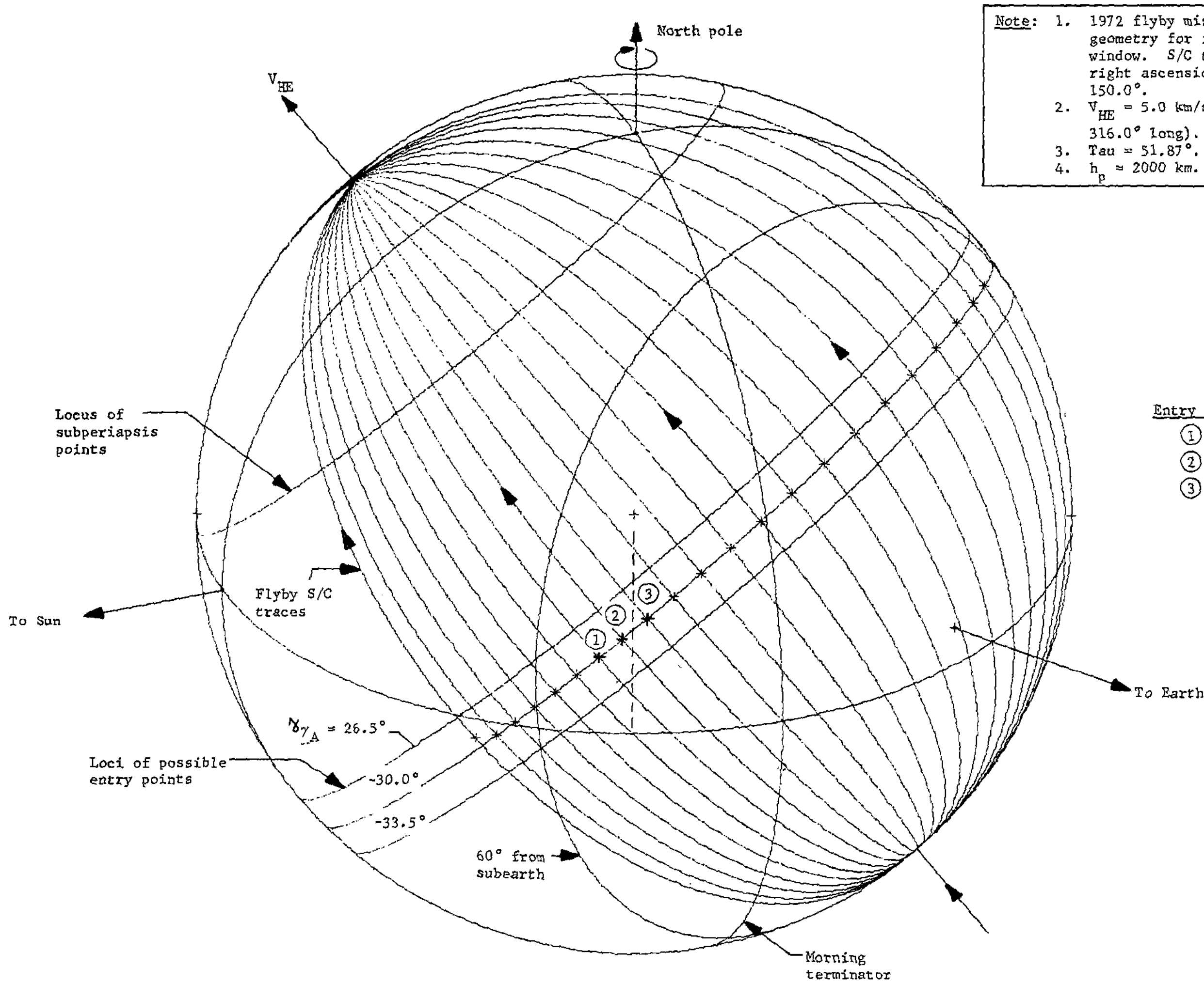
Mission Sequence

The overall flyby mission sequence is shown in figure 15 and the pertinent points are discussed below.

Prelaunch, launch, and cruise.- As in the orbital mission, the power is applied to the BVS and subsonic probe telemetry subsystems and the subsystems are monitored before liftoff. The capsule system is in a passive state except for status monitoring, battery charging, and electrical heaters for thermal control until 5 minutes before separation from the spacecraft.

Capsule initiation, separation, and stabilization.- Approximately 52 hr and 53 minutes before planet encounter the command is sent from Earth to the spacecraft and the capsule initiation phase begins. This phase through the capsule stabilization phase is identical to that of the orbital mission.

Capsule deflection, coast, and entry.- The capsule is given a velocity increment of 50 mps for this mission. The other associated events are identical to that of the orbital mission up to the time of atmospheric entry. The entry environments for this direct entry at 37 4000 fps are shown in figures 16, 17, and 18. The BVS deployment sequence and subsonic probe ejection are identical to that of the orbital mission.



Note:
1. 1972 flyby mission encounter geometry for first day of launch window. S/C traces shown for right ascension from 46.0° to 150.0°.
2. V_{HE} = 5.0 km/sec (at 45.0°N lat, 316.0° long).
3. Tau = 51.87°.
4. h_p = 2000 km.

Entry Points

1. 10.8° lat, 65.4° long.
2. 13.5° lat, 68.5° long.
3. 16.2° lat, 71.9° long.

Figure 13.- 1972 Targeting

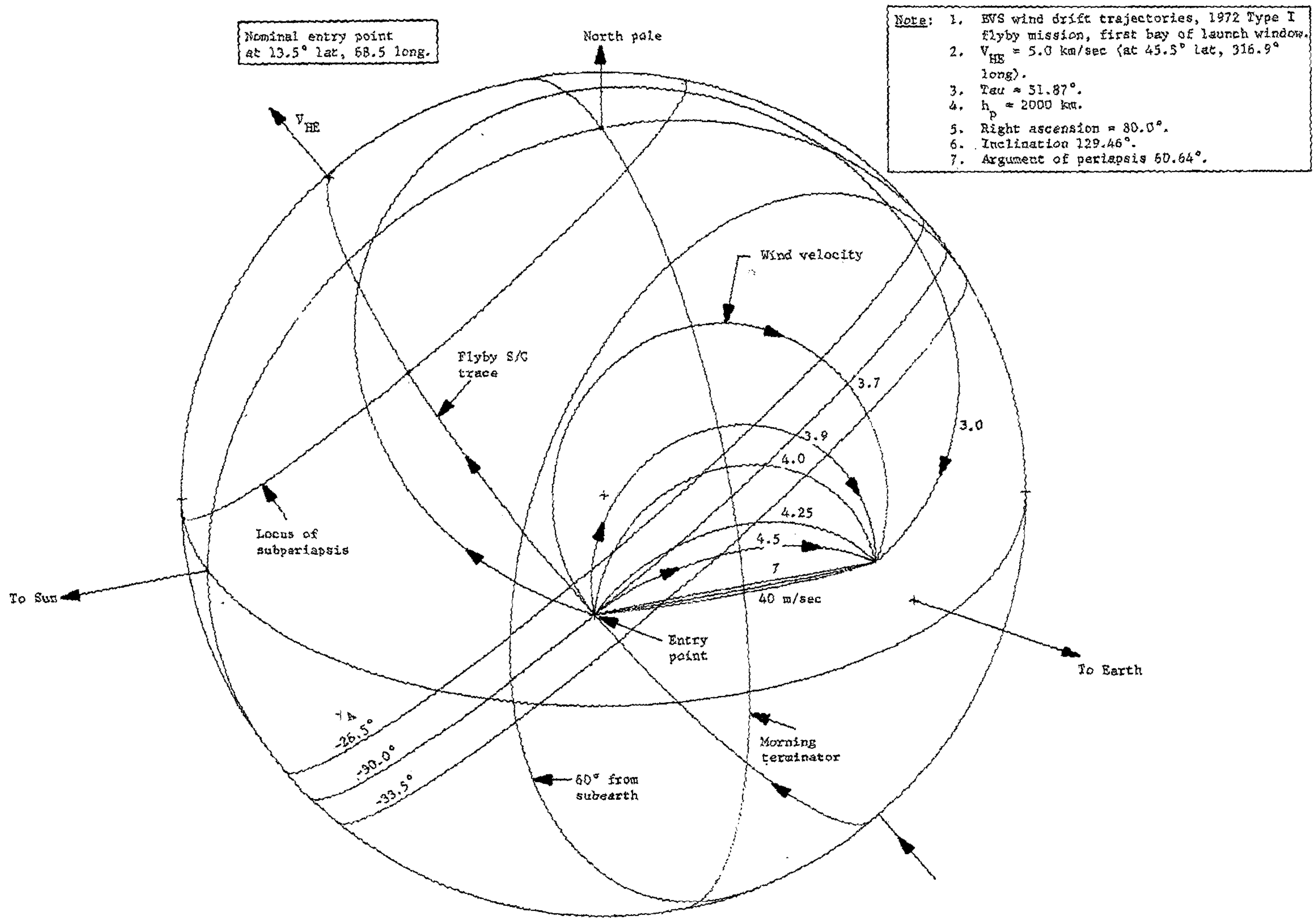


Figure 14.- EVS Wind Drift Trajectories

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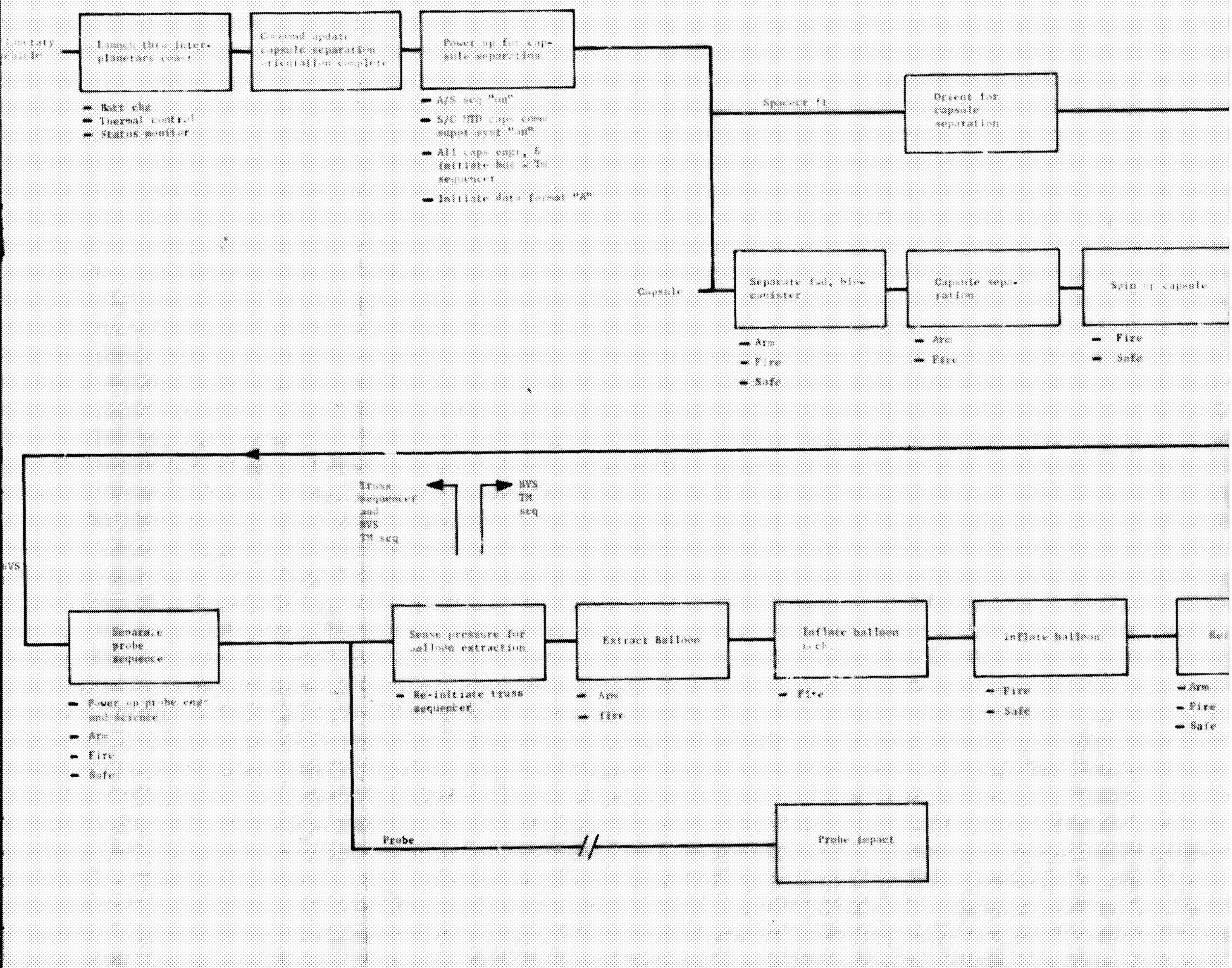

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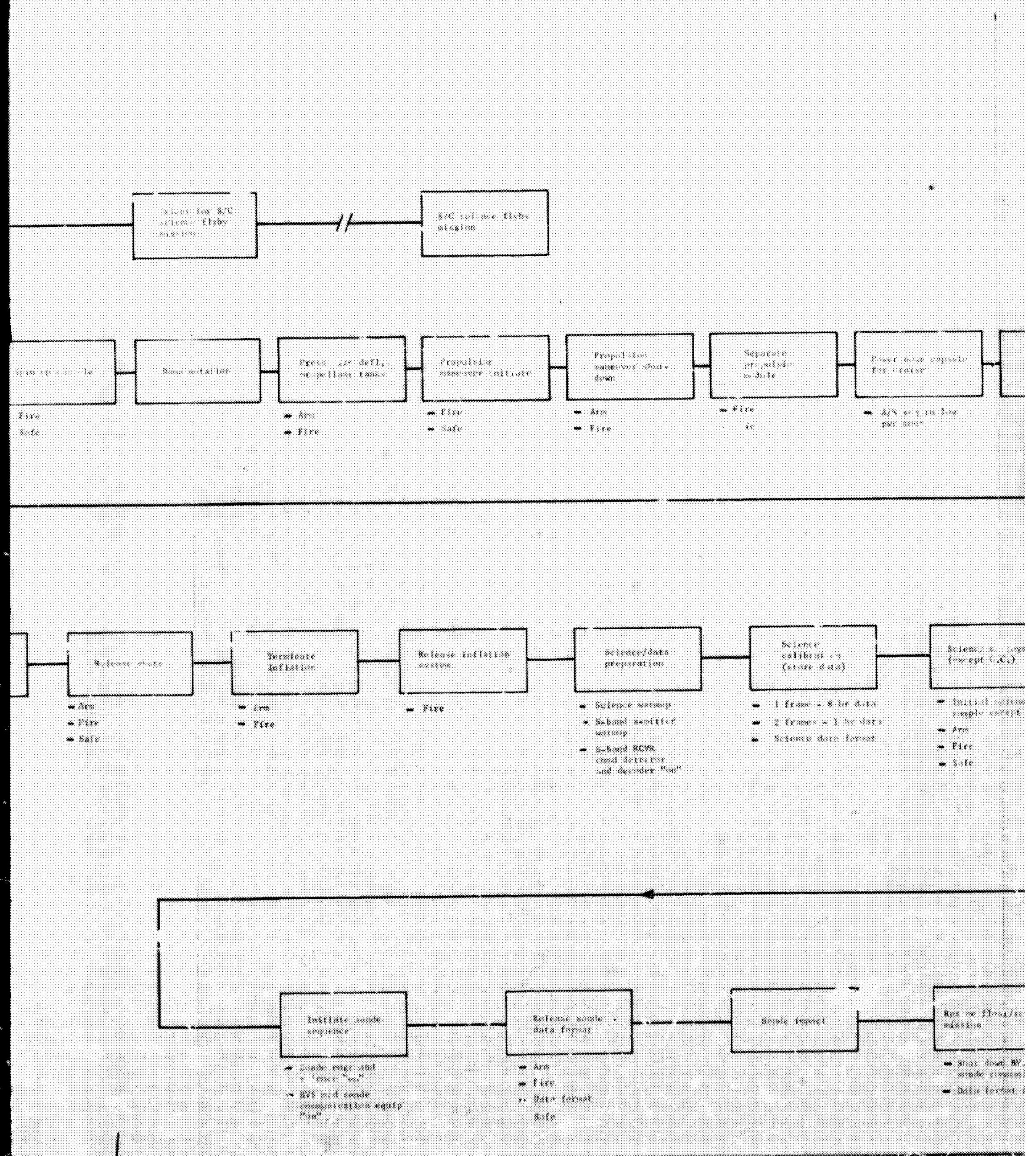
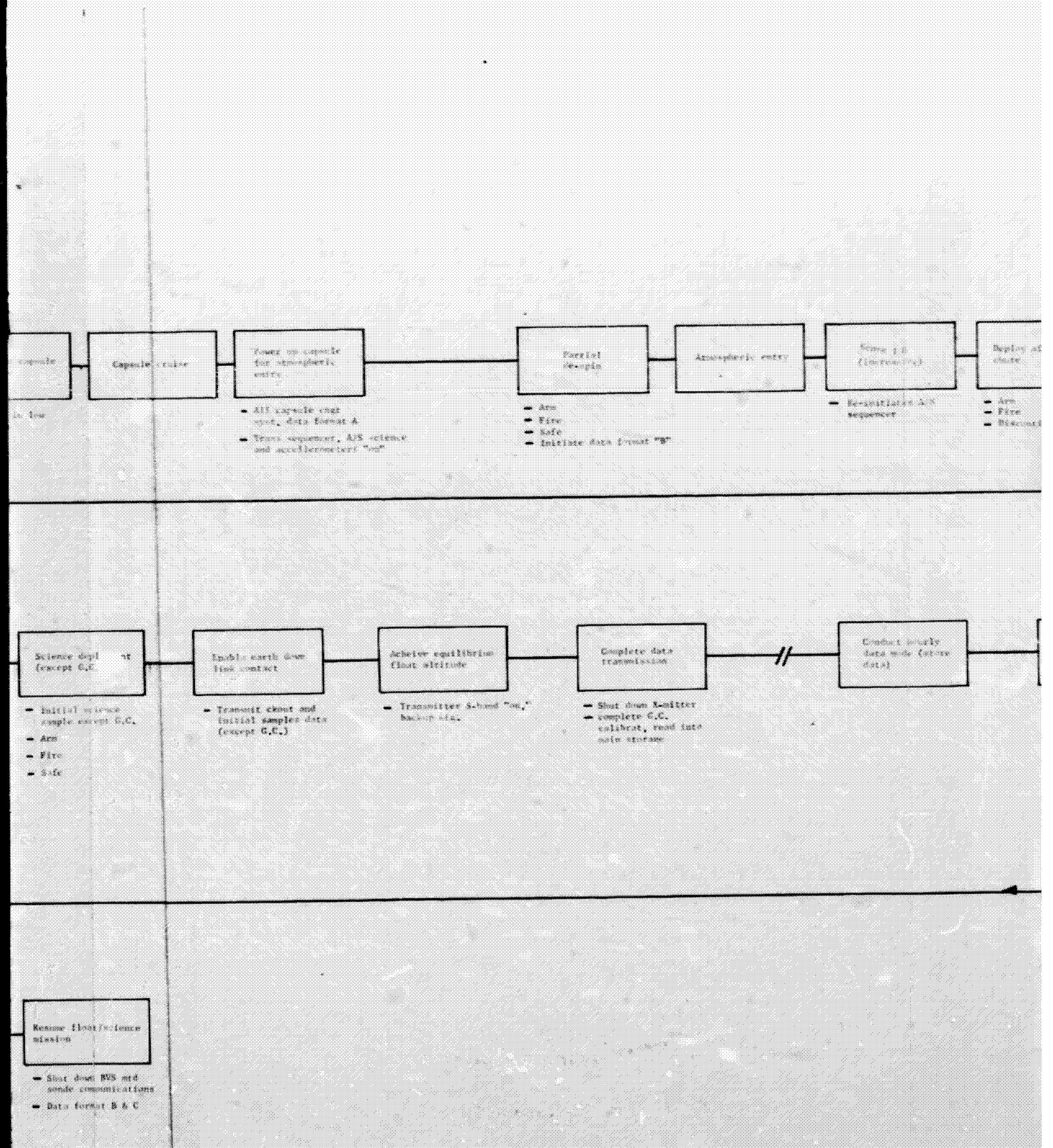


FOLDOUT FRAME 3



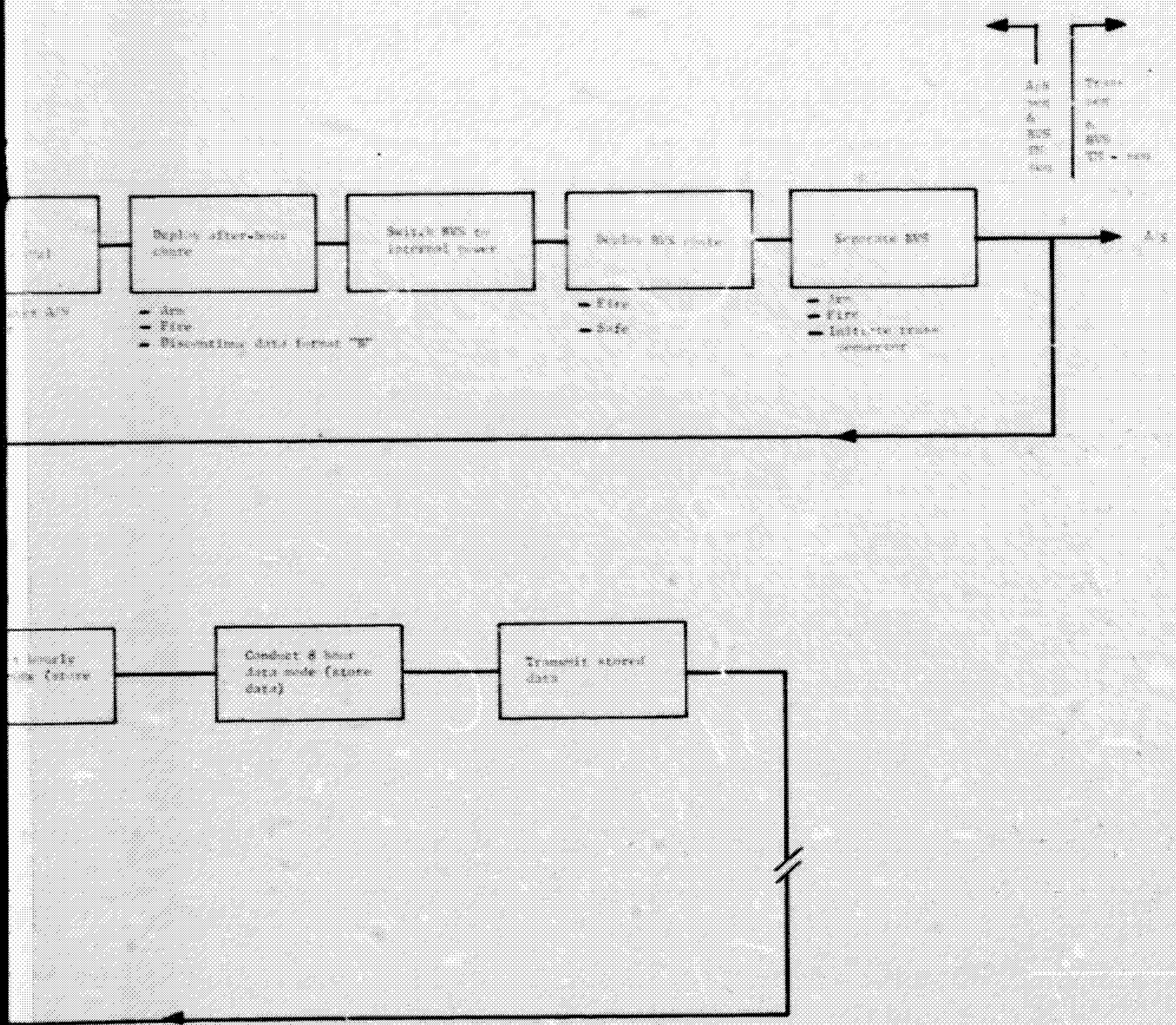


Figure 15.- Flyby Mission Sequence

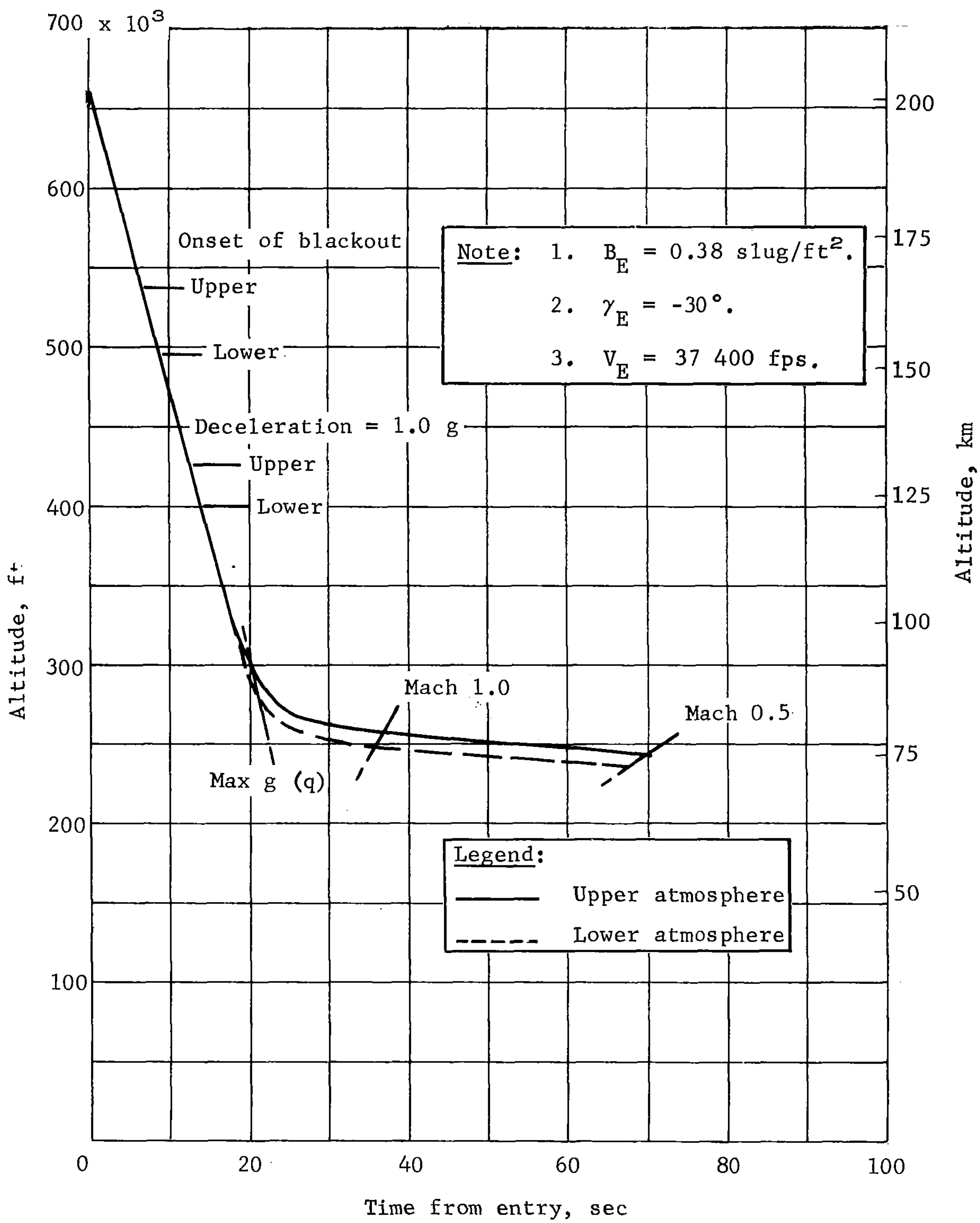


Figure 16.- BVS Reference Entry Trajectory, 1972 Flyby Mission, Altitude vs Time from Entry

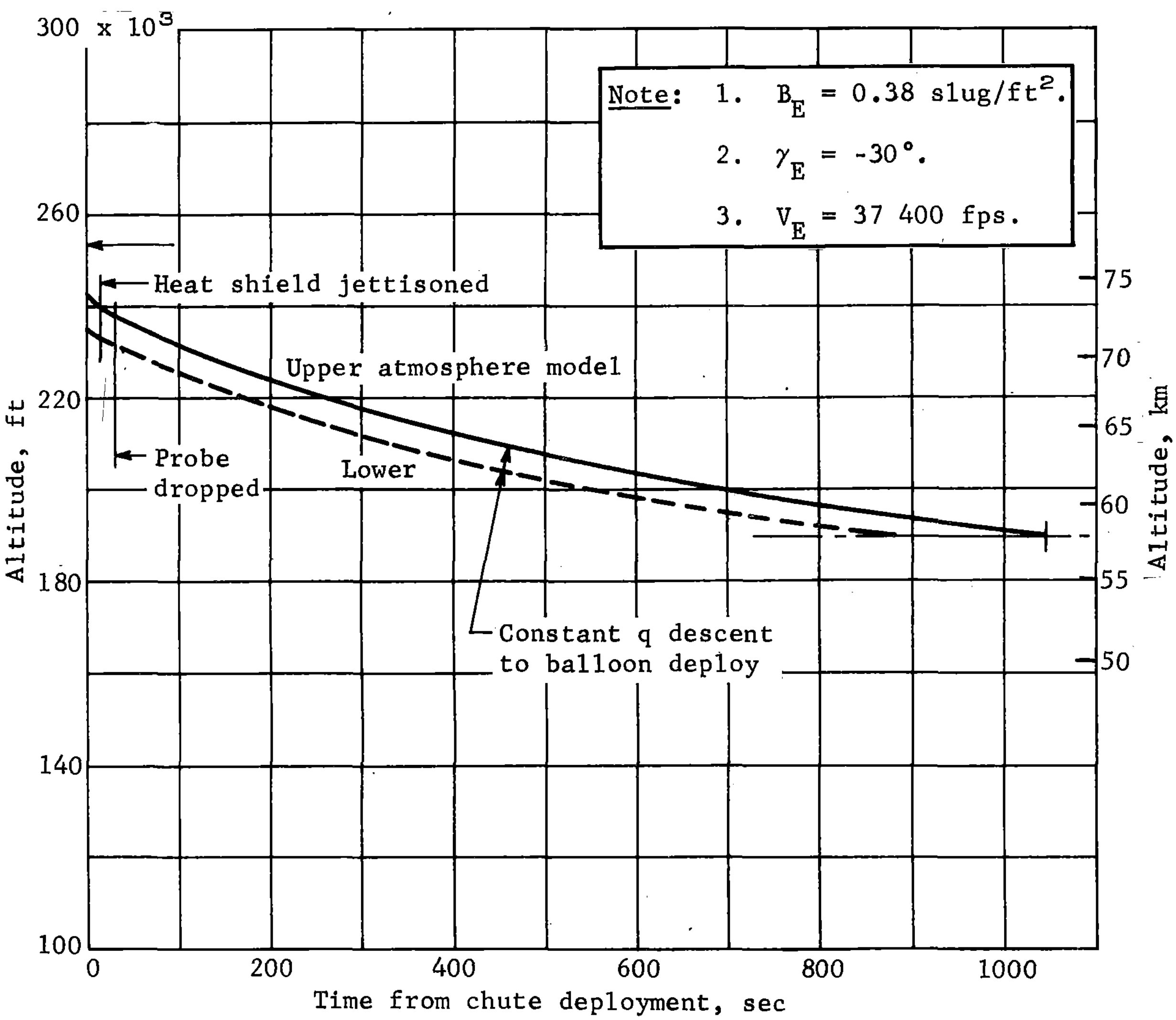


Figure 17.- BVS Reference Trajectory, 1972 Flyby Mission, Altitude vs Time from Deployment

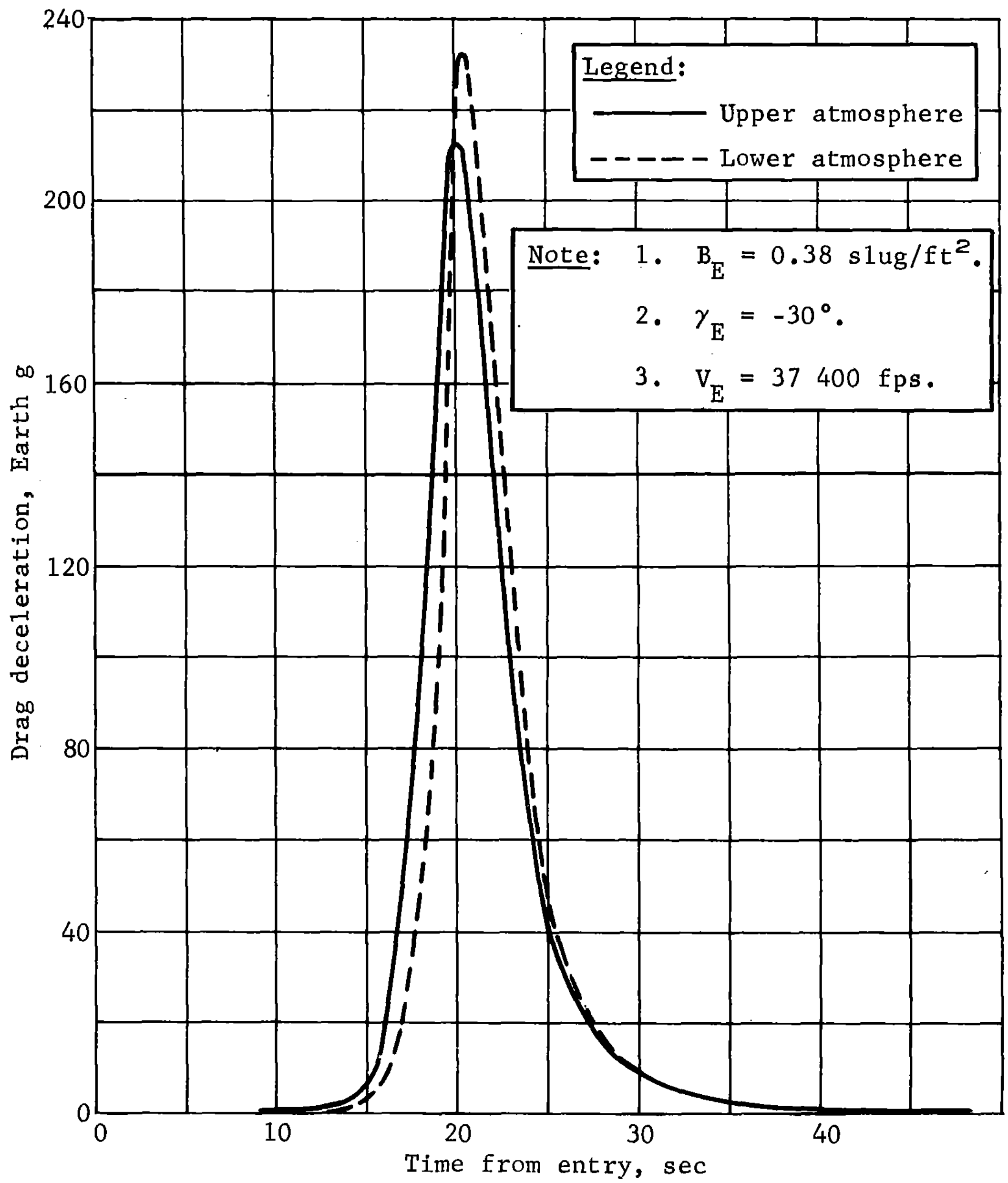


Figure 18.- BVS Reference Entry Trajectory, 1972 Flyby Mission, Drag Deceleration vs Time from Entry

TABLE 11.- BVS FLYBY MISSION WEIGHT STATEMENT

Launch vehicle (Titan IIIC Capability)				2425
Planetary vehicle at cruise				(2125)
Spacecraft			951	
S/C at planet		882		
BVS/entry vehicle system			1174	
Entry weight		963		
Mission margin				300
BVS/entry vehicle system				(1174)
Capsule/spacecraft adapter (including biocanister)			153	
Propulsion module (including spin design)			58	
BVS/entry vehicle at entry			(963)	
BVS		415		
Subsonic probe		85		
Aeroshell with equipment		463		
Aeroshell	382			

BVS flotation.- The BVS floats in the environment shown in table 6, but for this mission it transmits its data directly to earth following each 8-hr data sample period. On command from Earth one of the two drop sondes is released and descends to the surface relaying its data to the BVS.

A detail sequence of events is given in table 12.

General configuration.- The planetary vehicle shown in figure 19 in the cruise configuration, consists of the spacecraft and an entry capsule system. The flyby spacecraft is based on a modified Mariner '69.

TABLE 12.- 1972 VENUS MISSION SEQUENCE OF EVENTS, BVS, FLYBY SPACECRAFT

Event	Time	Samps/sec	From system	To system	Remarks
. . . . Planetary vehicle launch and cruise	~ 180 days				
. . . . Command update for capsule separation orientation complete	S-00:06:00		(Internal to spacecraft)		Capsule system passive
1. Initiate capsule system and turn on aeroshell sequencer	S-00:05:00		S/C	Power	
. . . . Turn on S/C mounted capsule comm support equipment	S-00:05:00		(Internal to spacecraft)		
2. Turn on capsule engineering S/S and uhf transmitter, initiate BVS-TM seq.	S-00:04:59	1/4	A/S sequencer	A/S power	
3. Initiate data Mode A	S-00:04:58	1/4	BVS-TM sequencer	BVS T/M	
4. Arm forward biocanister separation pyro	S-00:04:40	1/4	A/S sequencer	A/S pyro	
5. Fire forward biocanister separation pyro	S-00:04:39	1/4	A/S sequencer	A/S pyro	3 ignitors, shaped charge
6. Safe forward biocanister separation pyro	S-00:04:38	1/4	A/S sequencer	A/S pyro	
. . . . Orient spacecraft for capsule separation			(Internal to spacecraft)		
7. Arm capsule separation and spin pyro	S-00:00:01	1/4	A/S sequencer	A/S pyro	
8. Fire capsule separation pyro	S-00:00:00	2	A/S sequencer	A/S pyro	4 bolts
9. Fire capsule spin up pyro	S+00:00:01	2	A/S sequencer	A/S pyro	2 ignitors, solid rockets
10. Safe capsule separation and spin pyro	S+00:00:02	1/4	A/S sequencer	A/S pyro	
. . . . Orient spacecraft for Venus flyby			(Internal to spacecraft)		
11. Arm deflection propulsion pressurization and pyro start	S+00:19:49	1/4	A/S sequencer	A/S pyro	
12. Fire deflection propulsion pressurization pyro	S+00:19:50	1/4	A/S sequencer	A/S pyro	1 valve
13. Fire deflection propulsion start pyro	S+00:20:00	2	A/S sequencer	A/S pyro	1 valve
14. Safe deflection propulsion pressurization and start pyro	S+00:20:01	1/4	A/S sequencer	A/S pyro	
15. Arm deflection propulsion shutdown and separation pyro	S+00:21:33	1/4	A/S sequencer	A/S pyro	
16. Fire deflection propulsion shutdown pyro	S+00:21:34	2	A/S sequencer	A/S pyro	1 valve
17. Fire deflection propulsion separation pyro	S+00:21:39	1/4	A/S sequencer	A/S pyro	2 bolts
18. Safe deflection propulsion shutdown and separation pyro	S+00:21:40	1/4	A/S sequencer	A/S pyro	
19. Power down capsule for cruise	S+00:22:00	1/4	A/S sequencer	A/S power	Sequencer in low power mode
20. Power up capsule for atmospheric entry, entry, all Engr, data mode A	S+52:49:34	1/4	A/S & BVS TM seq's	A/S power	T/M uhf on, Format A
21. Turn on truss sequencer, aeroshell science, and accelerometer	S+52:49:35	1/4	A/S sequencer	A/S power	
22. Arm partial despin pyro	S+52:54:33	1/4	A/S sequencer	A/S pyro	
23. Fire partial despin pyro	S+52:54:34	1/4	A/S sequencer	A/S pyro	1 cable cutter
24. Safe partial despin pyro	S+52:54:35	1/4	A/S sequencer	A/S pyro	
25. Sense 1 g increasing (atmospheric entry) initiate data mode B	S+53:09:34 E-00:00:00	2	Guidance & Control & BVS TM sequencer	A/S sequencer	
26. Arm afterbody chute deployment pyro	E+00:00:55	1/4	A/S sequencer	A/S pyro	
27. Fire afterbody chute deployment pyro, discontinue data mode B	E+00:00:56	1	A/S sequencer	A/S pyro	1 (Mach 0.5)
28. Switch BVS to internal power	E+00:01:09	1/4	A/S sequencer	BVS power	
29. Fire afterbody deployment pyro	E+00:01:10	1	A/S sequencer	A/S pyro	3 bolts
30. Safe afterbody chute & afterbody deploy pyro	E+00:01:11	1/4	A/S sequencer	A/S pyro	
31. Arm separate BVS pyro	E+00:01:25	1/4	A/S sequencer	A/S pyro	
32. Fire separate BVS pyro, and initiate truss sequencer	E+00:01:26 ST-00:00:00	2	A/S sequencer	A/S pyro	2 bolts
33. Initiate probe (power up probe science & engr)	ST+00:00:01	1/4	Truss sequencer	Probe power	Power on probe engineering and science
34. Arm probe separation pyro	ST+00:00:14	1/4	Truss sequencer	Truss pyro	
35. Fire probe separation pyro	ST+00:00:15	1	Truss sequencer	Truss pyro	1 bolt
36. Safe probe separation pyro	ST+00:00:16	1/4	Truss sequencer	Truss pyro	
37. Initiate data playback transmit from storage	ST+00:01:00	1/4	BVS-TM sequencer	BVS-TM	
. . . . Probe impact	ST+00:36:00				S/C mounted equipment shutdown
38. Sense balloon extraction pressure	~ST+00:15:00 P-00:00:00	2	Pressure switch	Truss sequencer	
39. Arm balloon extraction and inflation pyro	P+00:00:01	1/4	Truss sequencer	Truss pyro	
40. Fire balloon extraction pyro	P+00:00:02	2	Truss sequencer	Truss pyro	2 bolts
41. Fire sock inflation pyro	P+00:00:03	2	Truss sequencer	Truss pyro	1 valve
42. Fire balloon inflation pyro	P+00:00:05	2	Truss sequencer	Truss pyro	1 valve
43. Safe balloon extraction and inflation pyro	P+00:00:06	1/4	Truss sequencer	Truss pyro	
44. Arm BVS chute release pyro	P+00:00:26	1/4	Truss sequencer	Truss pyro	
45. Fire BVS chute release pyro	P+00:00:27	1	Truss sequencer	Truss pyro	1 cable cutter
46. Safe BVS chute release pyro	P+00:00:28	1/4	Truss sequencer	Truss pyro	
47. Arm inflation termination and inflation system release pyro	P+00:00:58	1/4	Truss sequencer	Truss pyro	
48. Fire inflation termination pyro	P+00:00:59	2	Truss sequencer	Truss pyro	
49. Fire inflation system release pyro, re-initiate BVS TM sequencer	P+00:01:00	1/4	Truss sequencer	Truss pyro	3-1 tube cutter; 2 bolts
50. Science warm up, S-band X-mitter rovr, cmmd detector & decoder on	P+00:01:01	1/4	BVS-TM sequencer	BVS TM & power	
51. Science calibration (stored data), & data mode C & D	P+00:03:00	1/4	BVS-TM sequencer	BVS-science	
52. Arm science deployment pyro (except gas chromatograph)	P+00:03:29	1/4	BVS-TM sequencer	BVS-pyro	
53. Fire science deployment pyro, open M.S. solenoid valves gather initial science data sample	P+00:03:30	1/4	BVS-TM sequencer	BVS-pyro	1 cable cutter
54. Safe science deployment pyro	P+00:03:31	1/4	BVS-TM sequencer	BVS-pyro	
55. Enable Earth down link contact and transmit ckout and initial sample data (except gas chromatograph)	P+00:04:00	1/4	BVS-TM sequencer	BVS-TM	
56. Initiate backup signal for S-band X-mitter	P+00:08:00	1/4	BVS-TM sequencer	BVS-TM	
57. Shut down S-band X-mitter, read G.C. data into main storage	P+00:24:00	1/4	BVS-TM sequencer	BVS-TM	
58. Initiate hourly data sample Mode C - store data	P+01:00:00 & every hour thereafter	1/4	BVS-TM sequencer	BVS-TM	
59. Initiate 8 hour data sample Mode D	P+06:00:00 & every 8 hr thereafter		BVS-TM sequencer	BVS-TM	
60. Transmit stored data	P+08:00:00 & every 8 hr thereafter		BVS-TM sequencer	BVS-TM	
. . . . Initiate sonde sequence	P+ - - -				
61. Turn on sonde engineering and science and turn on BVS mounted sonde communication equipment	SR-00:00:00 SR+00:00:01	1/4	Earth command BVS T/M sequencer	BVS-TM Sonde power	Repeat for sonde 2
62. Arm sonde release pyro	SR+00:00:59	1/4	BVS T/M sequencer	BVS pyro	
63. Fire sonde release pyro and initiate data format	SR+00:01:00	1/4	BVS T/M sequencer	BVS pyro	1 cable cutter
64. Safe sonde release pyro	SR+00:01:01	1/4	BVS T/M sequencer	BVS pyro	
. . . . Sonde impact	EST SR+00:39:00	1			
65. Shutdown BVS mounted sonde communication equipment	SR+00:40:00	1/4	BVS T/M sequencer	BVS power	

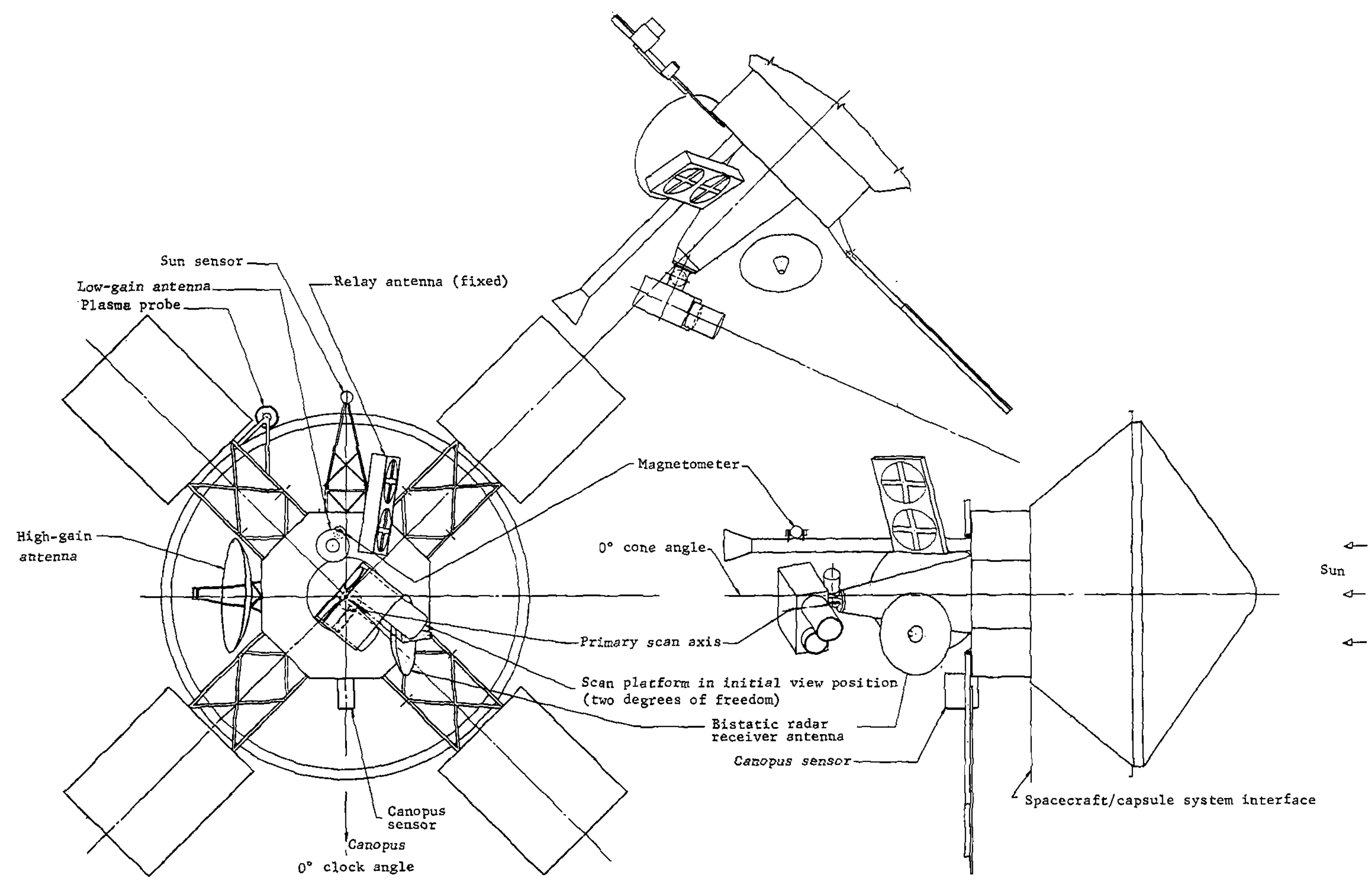


Figure 19.- Planetary Vehicle for Flyby Mission

VENUS/MERCURY MISSION

The statement of work specified a mission to be considered with a spacecraft swingby of Venus in which a BVS/entry vehicle is separated for Venus entry before the S/C flyby of Mercury. Three options to the mission were to be investigated: (1) ballistic mission of the spacecraft with the BVS/entry vehicle entering the Venus atmosphere without retardation (entry velocity of 43 550 fps); (2) ballistic mission of the S/C with retardation of the BVS/entry vehicle just before atmospheric entry (entry velocity reduced to 38 000 fps); and (3) S/C impulse in the vicinity of Venus, allowing for lower energy on approach to Venus (entry velocity of 40 700 fps). These three options are summarized in table 13. The all-ballistic mission, option (1) above, was selected for further definition and documentation. This decision was based primarily on the desire to study the significance of the extreme entry velocity on the BVS/entry vehicle.

The retardation of the capsule before entry is feasible and presents an entry environment identical to that studied for the 1972 flyby mission. The S/C impulse mode allows for the intermediate entry environment, but at the expense of a large propulsion module attached to the S/C and the requirement of assisting the upper stage of the Titan IIIC booster to attain injection into the heliocentric orbit. The guidelines under which the baseline mission is defined are given in table 14.

The major points of difference between the three mission modes are atmospheric entry velocity of the BVS/entry vehicle, requirement for a propulsion module to retard the capsule to produce the lower entry velocity, and requirement for a S/C propulsion module to produce the impulse at Venus which also results in a lower BVS/entry vehicle entry velocity.

Mission Description

The baseline mission shown in figure 20 has a Type I transfer trajectory on a swingby of Venus to a flyby of Mercury with a launch period of 25 October thru 7 November 1973. Approximately 68 hr before Venus encounter the BVS/entry vehicle is separated and deflected for entry at Venus. The S/C continues on a swingby of Venus. For this mission the S/C does not relay BVS or subsonic probe data; all of these data are transmitted directly to Earth. The mission is summarized in table 15.

TABLE 13.- VENUS/MERCURY FLYBY MISSION OPTION COMPARISON

Item	All-ballistic mission	S/C impulse at Venus	Capsule Retro at Venus
Launch period	Oct 25 - Nov 7, 1973	Oct 11 - Oct 21, 1973	Oct 25 - Nov 7, 1973
Arrival date	Feb 5, 1974	Feb 9, 1974	Feb 5, 1974
Target for entry	~41° from subearth	~45° from subearth	~41° from subearth
C_3/V_{HE}	19.5/8.5	19.5/7.1	19.5/8;5
Entry velocity, fps	43,550	40.700	38,000
Heatshield weight, lb	[a]270	210	[b]235
BVS mission	Dark side	Dark side	Dark side
Capsule propulsion	Deflection only with ΔV = 93 mps	Deflection only with ΔV = 93 mps	Deflection and retro ΔV = 1700 mps
Capsule propulsion weight, lb	83	80	1200 - 1500
Capsule ACS	3-axis control, small thrusters	3-axis control, small thrusters	3-axis control, large thrusters
S/C propulsion	Midcourse only	Injection, midcourse and impulse maneuver ΔV = 3.34 km/sec	Injection and midcourse ΔV = 1.65 km/sec

[a]7.0-ft-diam aeroshell.

[b]8.5-ft-diam aeroshell.

TABLE 14.- VENUS/MERCURY MISSION GUIDELINES

Mission mode	Flyby of Mercury with swingby of Venus
Launch opportunity	1973
Launch vehicle	Titan IIIC
Spacecraft	Modified Mariner as defined by JPL Document 760-1
Buoyant station	Modified minimum weight BVS investigated under Contract NAS1-6607
External entry shape	Large angle blunt cone
Communications	Relay via S/C or direct to Earth. Assume a full complement of Deep Space Net Stations are available for support of mission

TABLE 15.- 1973 VENUS/MERCURY MISSION SUMMARY

Flyby spacecraft (modified Mariner)
- 1049 lb at Venus flyby
- $H_p = 5505$ km

Entry vehicle
- $V_E = 43\ 550$ fps
- $\gamma_e \doteq -35°$
- $B_E = 0.58$ slugs/ft^2
- 68-hr coast period
- Deflection impulse of 93 mps
- Radius at ejection = 2 000 000 km

400-lb BVS
- In-the-cloud mission (R = 6108 km)
  - $P_A = 612$ mb (~9 psia)
  - $T_A = 70°F$
- Deployed approximately 41° from subearth
- 58 lb of science (175-lb gondola)
  - Minibio laboratory
  - Atmospheric measurements
  - Radar altimeter
- Direct link to Earth (120 bps)
- Hydrogen inflated superpressure balloon
  - 4 manifolded tanks
  - Blowdown inflation
  - Supported by parachute during inflation

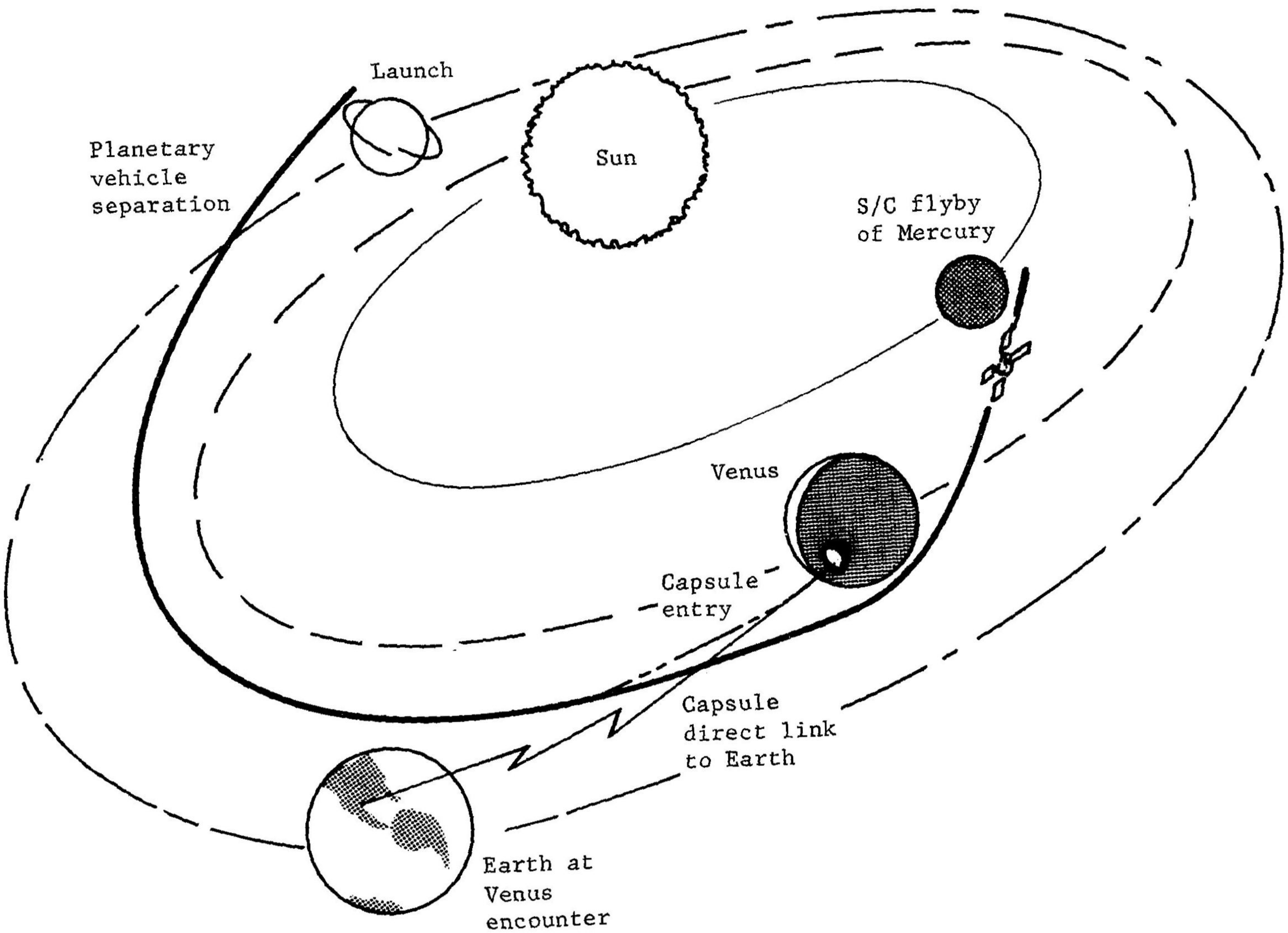


Figure 20.- 1973 Venus/Mercury Mission Profile for BVS

The entry site was selected to be compatible with the constraints of a -35° entry angle, not more than 60° from subearth for communications, and at least 20° from the terminator to allow an adequate mission before corssing the terminator with the balloon. Wind patterns were the main factor in selecting a point within the area defined by the above parameters. With wind velocities from 4 to 40 mps, the BVS drift remains within the 60° from subearth constraint for several hours or possible days. Figure 21 shows the entry point selected (41° from subearth) on the dark side of the planet. Figure 22 shows the effect of wind velocities within the model for the selected entry location.

The aeroshell for this mission has been reduced in size from the 8.5 ft diameter to a 7.0 ft diameter. This is necessitated by the heat shield weight allocation required for the high entry velocity of 43 550 fps. While the flyby and orbital missions could use a lightweight elastomeric heat shield material, this mission appears to require use of a carbon phenolic-type material with its attendant high density and weight.

An active three-axis attitude control system is used for this mission in place of the simple, spin-stabilized system used for the orbital and flyby missions. The active system is required to reorient the capsule following the deflection maneuver. After this reorientation the capsule is spun-up for the 68-hr coast period and atmospheric entry is accomplished in the spin mode.

The overall mission weight summary is given in Table 16, which indicates the 135-lb margin for the Titan IIIC launch vehicle. Additional detail on this mission is given in Volume II of this report.

Mission Sequence

The overall mission sequence is shown in figure 23 and the pertinent points are discussed below.

Prelaunch, launch, and cruise.- As in the orbital mission, power is applied to the BVS and subsonic probe telemetry for subsystem verification. The capsule system is in a passive state during cruise except for periodic status monitoring, battery charging, and thermal control until 5 minutes before separation from the S/C.

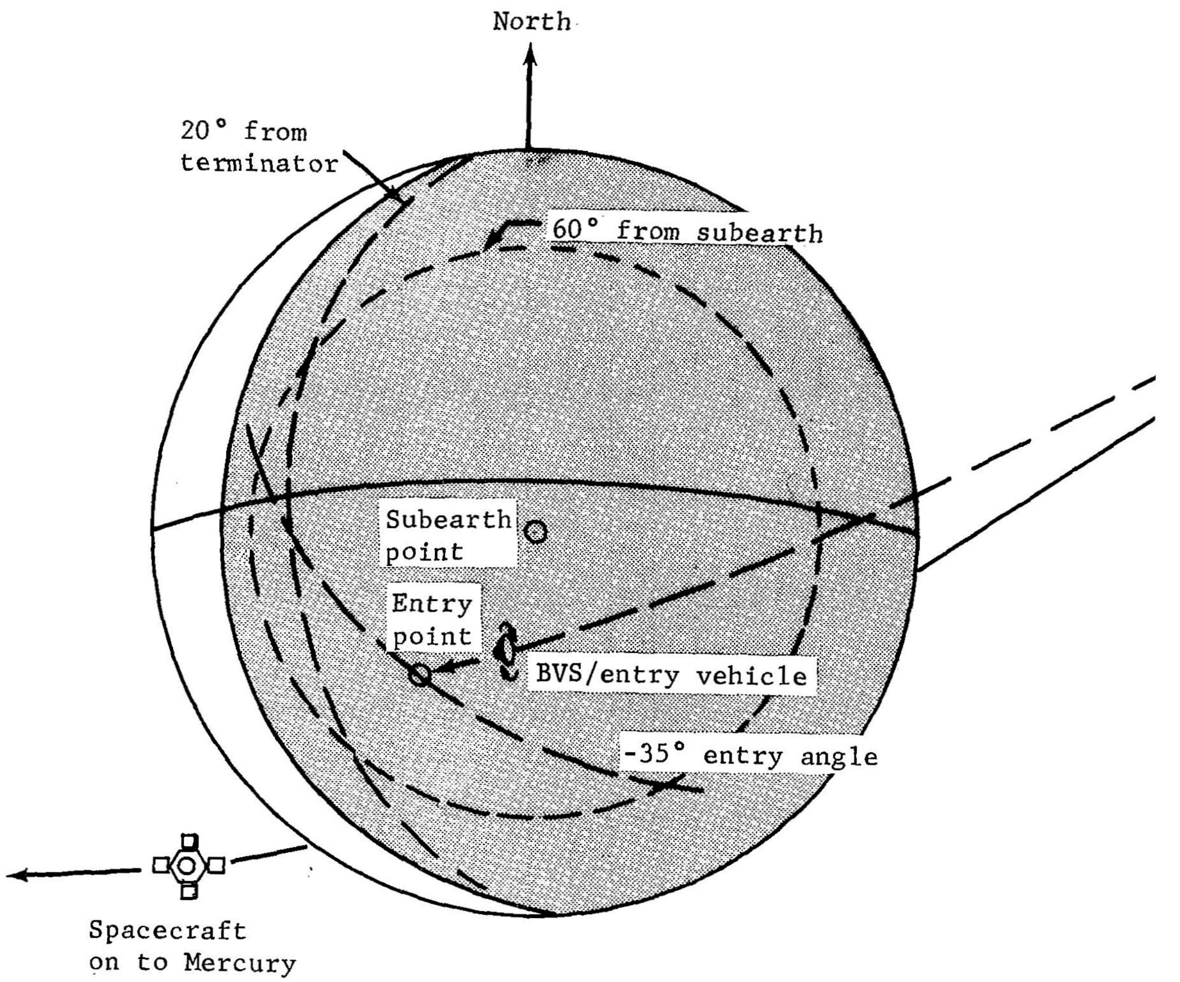


Figure 21.- BVS/Entry Vehicle Entry as Seen from Earth

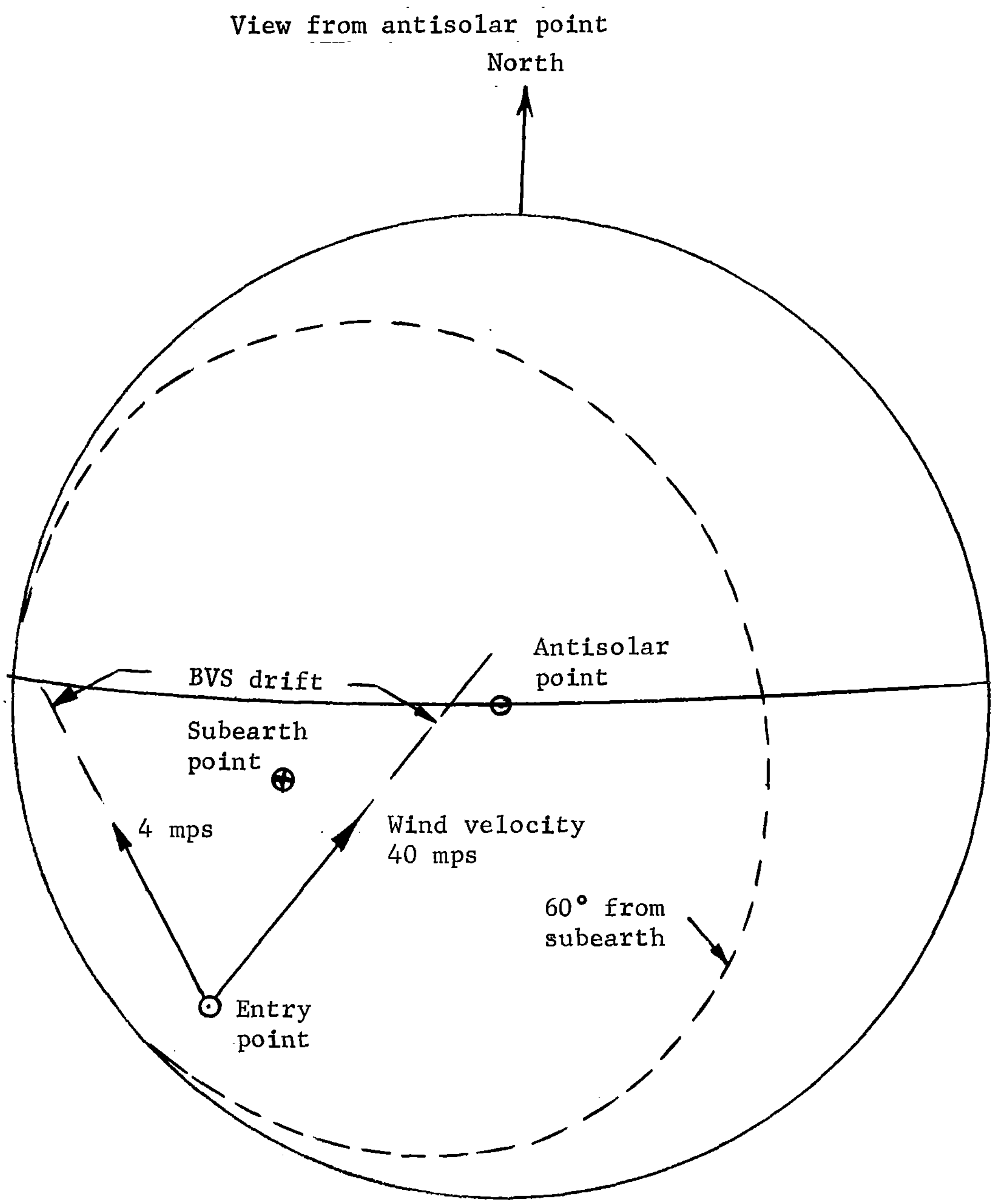
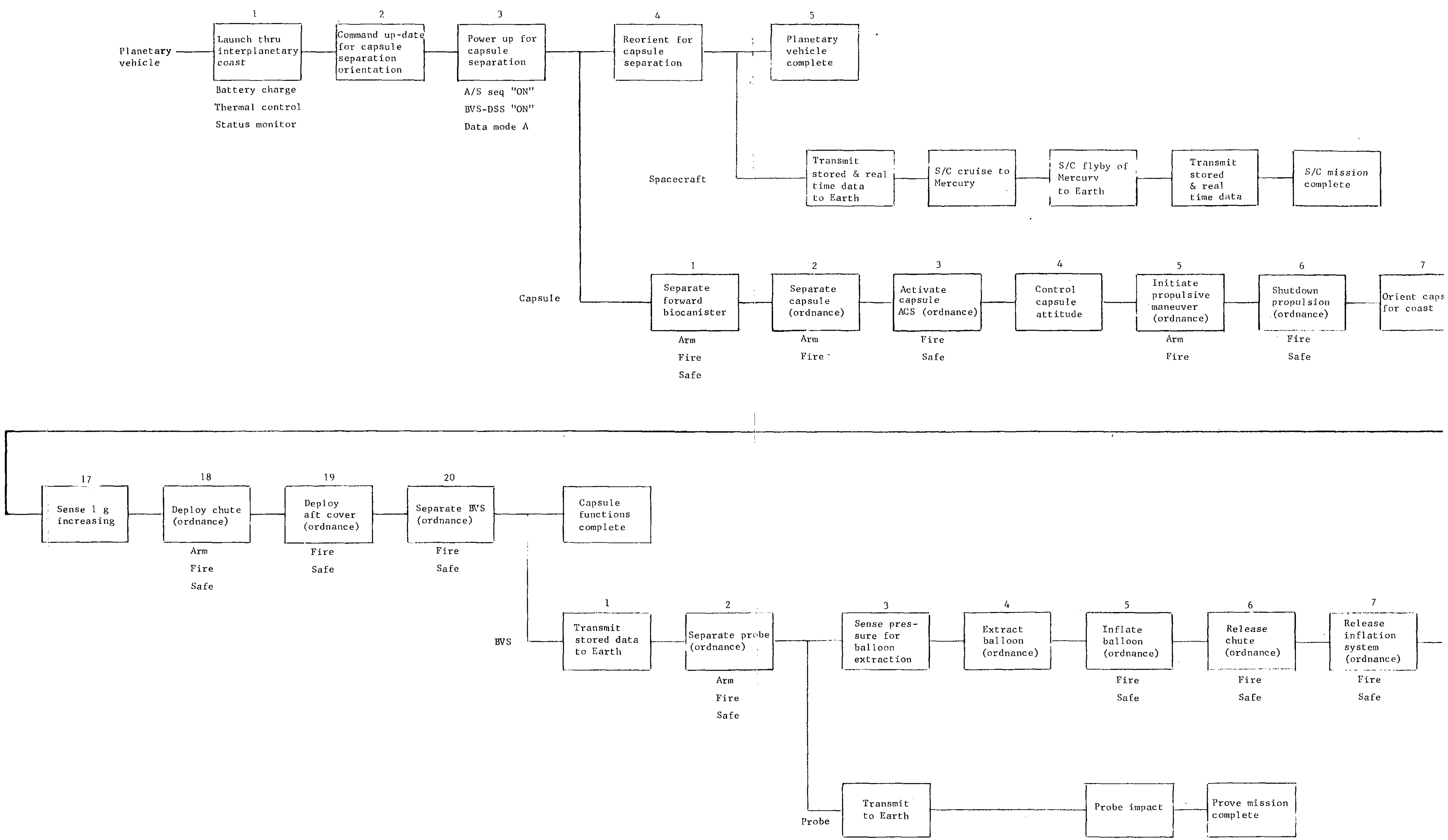


Figure 22.- BVS Drift Corridor for Range of Wind Velocities,
Dark Side Mission

TABLE 16.- BVS WITH VENUS/MERCURY FLYBY MISSION WEIGHT STATEMENT

Launch vehicle (Titan IIIC) capability			2500
Planetary vehicle at cruise			(2365)
Spacecraft		1115	
BVS/entry vehicle system		1250	
Entry weight	999		
Mission margin			135
BVS/entry vehicle system			(1250)
Capsule/spacecraft adapter (including biocanister)		113	
Propulsion module (including ACS)		138	
BVS/EV at entry		(999)	
(B = 0.58 slugs/ft^2)			
BVS	412		
Subsonic probe	85		
Aeroshell (7-ft diam)	444		
Aeroshell mounted equipment	58		



Planetary vehicle
1
Launch thru interplanetary coast
Battery charge
Thermal control
Status monitor
2
Command up-date for capsule separation orientation
3
Power up for capsule separation
A/S seq "ON"
BVS-DSS "ON"
Data mode A
4
Reorient for capsule separation
5
Planetary vehicle complete
Spacecraft
Transmit stored & real time data to Earth
S/C cruise to Mercury
S/C flyby of Mercury to Earth
Transmit stored & real time data
S/C mission complete
Capsule
1
Separate forward biocanister
Arm
Fire
Safe
2
Separate capsule (ordnance)
Arm
Fire
3
Activate capsule ACS (ordnance)
Fire
Safe
4
Control capsule attitude
5
Initiate propulsive maneuver (ordnance)
Arm
Fire
6
Shutdown propulsion (ordnance)
Fire
Safe
7
Orient caps for coast
17
Sense 1 g increasing
18
Deploy chute (ordnance)
Arm
Fire
Safe
19
Deploy aft cover (ordnance)
Fire
Safe
20
Separate BVS (ordnance)
Fire
Safe
Capsule functions complete
BVS
1
Transmit stored data to Earth
2
Separate probe (ordnance)
Arm
Fire
Safe
3
Sense pressure for balloon extraction
4
Extract balloon (ordnance)
5
Inflate balloon (ordnance)
Fire
Safe
6
Release chute (ordnance)
Fire
Safe
7
Release inflation system (ordnance)
Fire
Safe
Probe
Transmit to Earth
Probe impact
Prove mission complete

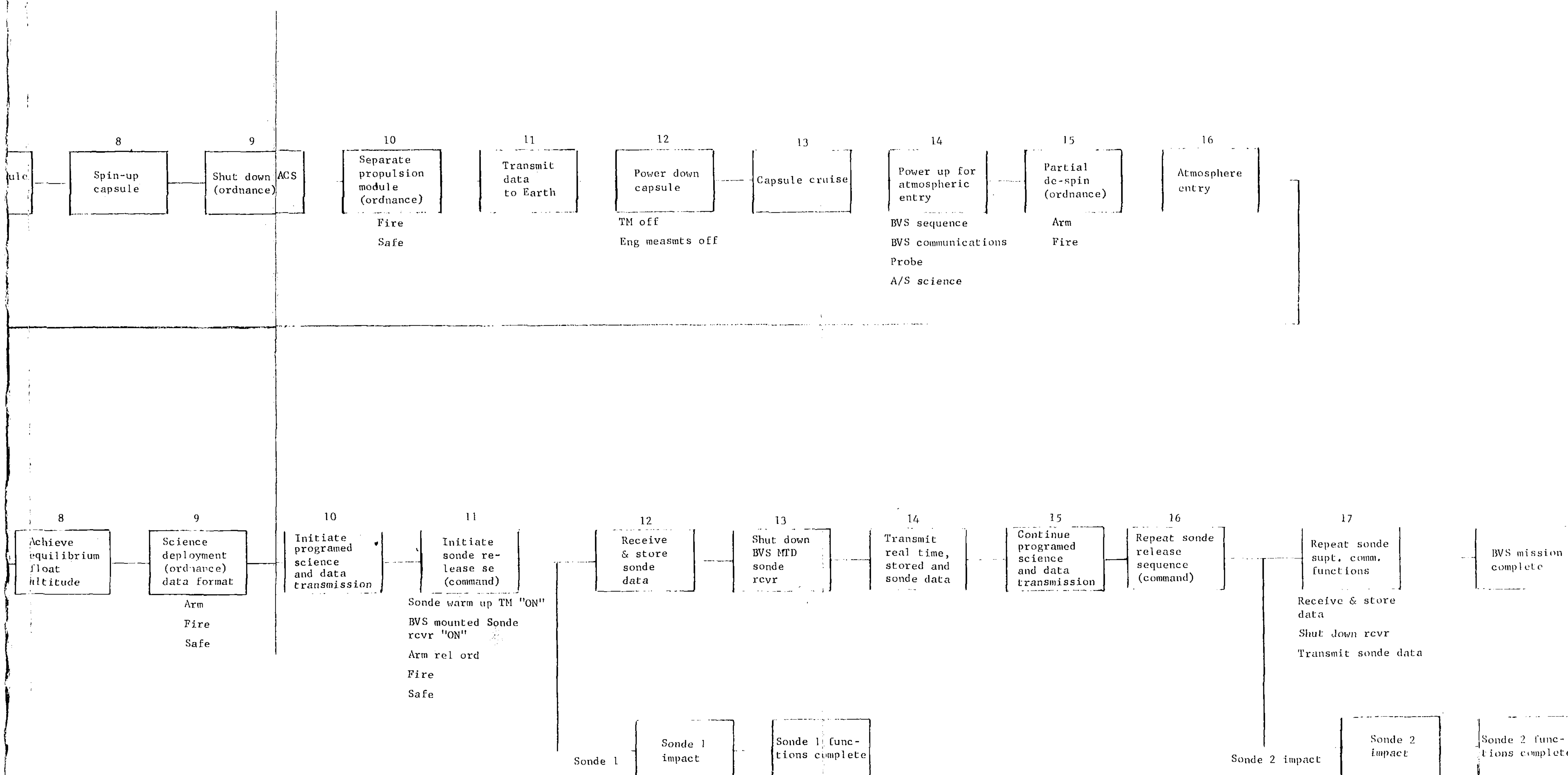


Figure 23.- Venus/Mercury Mission Sequence

Capsule initiation.- Approximately 68-hr before planet encounter the command is sent from earth to the S/C and the capsule initiation phase begins. This phase is identical through capsule separation to that of the orbital mission.

Capsule stabilization and deflection.- The capsule is controlled by a three-axis control system throughout the deflection impulse. The capsule is then oriented for proper attitude (at atmospheric entry) and spun up as for the orbital mission for the remainder of the coast period. The telemetry data stored during the separation, deflection, orientation, and spin-up is transmitted to earth.

Capsule coast and entry.- The capsule is powered down during the coast period. The sequence of events just before entry are identical to that of the orbital mission. The entry environments for this darkside mission entry velocity of 43 550 fps are shown in figures 24, 25, and 26. The BVS deployment sequence and subsonic probe ejection are identical to that of the orbital mission. However, for this mission the BVS and subsonic probe each transmit their data directly to Earth.

BVS flotation.- This phase of the mission is identical to that of the flyby mission. A detail sequence of events is given in table 17.

General Configuration.- The planetary vehicle shown in figure 19 in the cruise configuration, consists of a modified Mariner and the 7-ft-diam BVS/entry vehicle.

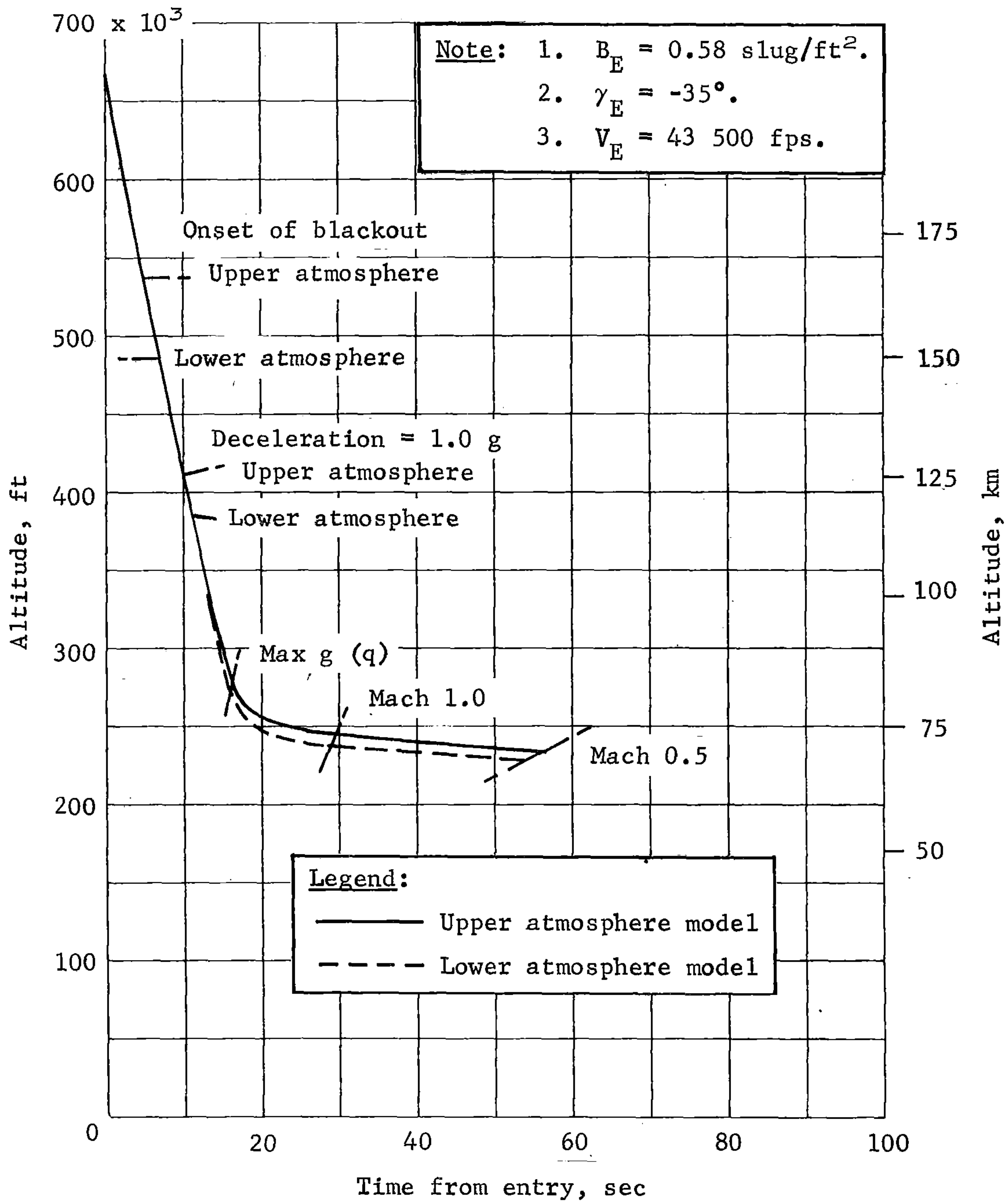


Figure 24.- BVS Reference Entry Trajectory, 1973 Venus/Mercury Flyby Mission

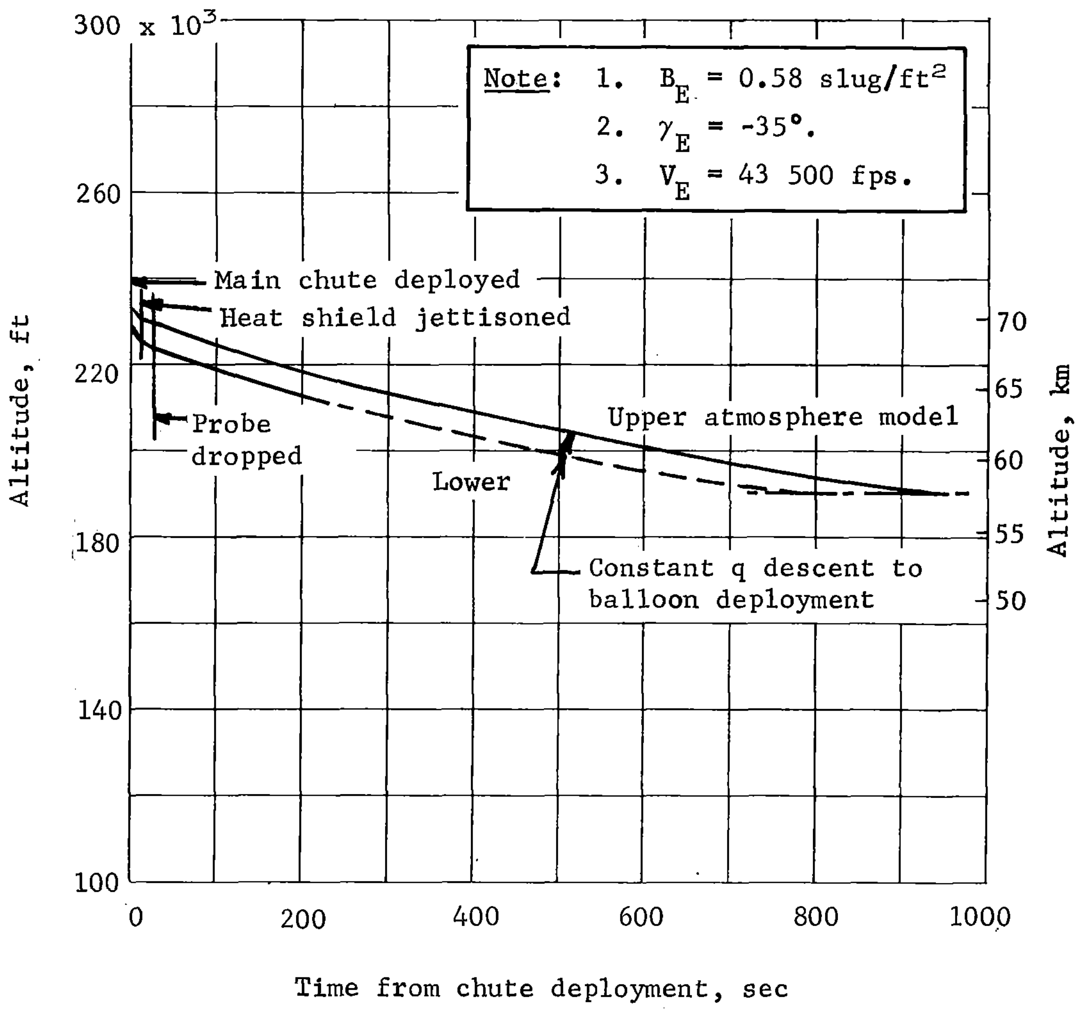


Figure 25.- BVS Reference Trajectory

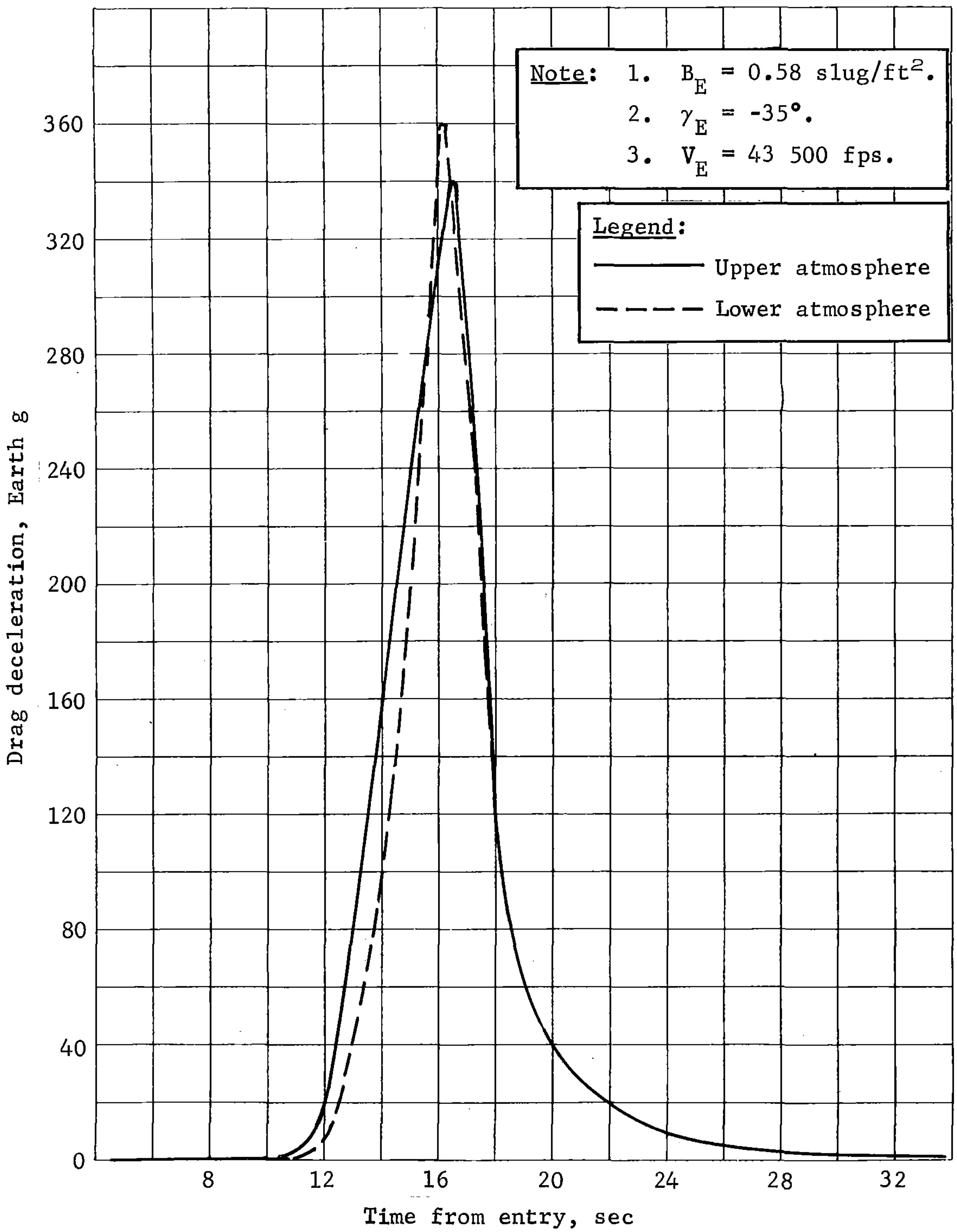


Figure 26.- BVS Reference Entry Trajectory, 1973 Venus/Mercury Flyby Mission

TABLE 17.- 1973 VENUS/MERCURY MISSION SEQUENCE OF EVENTS

Event	Time	Samps/ sec	From system	To system	Remarks
. . . . Planetary vehicle launch and cruise	105 days				
. . . . Command update for capsule separation					
. . . . Orientation complete	S-00:06:00		(Internal	to S/C)	Capsule system passive
1. Initiate capsule system & turn on aeroshell sequencer	S-00:05:00		S/C	A/S Power	
2. Turn on capsule engineering S/S, initiate BVS-DSS	S-00:04:59	1/16	A/S Seq	A/S Power	
3. Initiate data Mode A (stored data)	S-00:04:58	1/16	[a]BVS-DSS	[b]BVS-DS	
4. Arm forward biocanister separation pyro	S-00:04:40	1/16	A/S Seq	A/S Pyro	
5. Fire forward biocanister separation pyro	S-00:04:39	1/16	A/S Seq	A/S Pyro	3 Ignitors, shaped charge
6. Safe forward biocanister separation pyro	S-00:04:38	1/16	A/S Seq	A/S Pyro	
. . . . Orient S/C for capsule separation			(Internal	to S/C)	
7. Arm capsule separation and ACS pryo	S-00:00:01	1/16	A/S Seq	A/S Pyro	
8. Fire capsule separation pyro	S-00:00:00	1/2	A/S Seq	A/S Pyro	4 bolts
9. Fire capsule ACS pyro	S+00:00:01	1/2	A/S Seq	A/S Pyro	1 valve
10. Safe capsule separating and ACS pyro	S+00:00:02	1/16	A/S Seq	A/S Pyro	
. . . . Orient S/C for Venus flyby to Mercury			(Internal	to S/C)	
11. Arm deflection propulsion pressurization and start pyro	S+00:07:49	1/16	A/S Seq	A/S Pyro	
12. Fire deflection propulsion pressurization pyro	S+00:07:50	1/16	A/S Seq	A/S Pyro	1 valve
13. Fire deflection propulsion start pyro	S+00:08:00	1/2	A/S Seq	A/S Pyro	1 valve
14. Safe deflection propulsion pressurization and start pyro	S+00:08:01	1/16	A/S Seq	A/S Pyro	
15. Arm deflection propulsion shutdown and separation pyro	S+00:13:06	1/16	A/S Seq	A/S Pyro	
16. Fire deflection propulsion shutdown pyro	S+00:13:07	1/2	A/S Seq	A/S Pyro	1 valve
17. Fire deflection propulsion separation pyro	S+00:13:12	1/16	A/S Seq	A/S Pyro	2 bolts
18. Safe deflection propulsion shutdown and separation pyro	S+00:13:13	1/16	A/S Seq	A/S Pyro	
19. Orient capsule for spinup	S+00:13:15	1/2	A/S Seq	A/S G & C	
20. Command capsule ACS to spinup capsule	S+00:14:55	1/2	A/S Seq	A/S G & C	
21. Terminate ACS	S+00:14:58	1/2	A/S Seq	A/S Pyro	
22. Initiate data mode A (transmission)	S+00:15:00	1/16	BVS-DSS	BVS-DS	
23. Power down capsule for cruise	S+00:35:00	1/16	A/S Seq	A/S Power	Sequencer in low power mode
24. Power up for entry, all engr., data mode A (transmission)	S+67:53:00	1/16	A/S Seq	A/S Power	format A
25. Turn-on truss seq. A/S science and accelerometer	S+67:53:01	1/16	A/S Seq	A/S Power	
26. Arm partial despin pyro	S+67:57:50	1/16	A/S Seq	A/S Pyro	
27. Fire partial despin pyro	S+67:57:59	1/16	A/S Seq	A/S Pyro	1 cable cutter
28. Safe partial despin pyro	S+67:58:00	1/16	A/S Seq	A/S Pyro	
29. Sense 1.0 g increasing (atmos entry) initiate data Mode B (storage)	S+68:13:20 E-00:00:00	2	Accelero- meter	A/S Seq	Continue Mode A transmission
30. Fire afterbody, BVS chute deployment pyro	E+00:01:10	1	A/S Seq	A/S Seq	3 bolts
31. Safe afterbody chute & afterbody/BVS chute deploy pyro	E+00:01:11	1/4	A/S Seq	A/S Pyro	
32. Arm separate BVS pyro, power up BVS	E+00:01:25	1/4	A/S Seq	A/S Pyro & BVS-DSS	
33. Fire separate BVS pyro, and initiate truss sequencer	E+00:01:26 ST-00:00:00	2	A/S Seq	A/S Pyro	2 bolts
34. Initiate probe (power up probe science & engr)	ST+00:00:01	1/4	Truss Seq	Probe Power	Power on probe engr. & science
35. Arm probe separation pyro	ST+00:00 14	1/4	Truss Seq	Truss Pyro	
36. Fire prove separation pyro	ST+00:00:15	1	Truss Seq	Truss Pyro	1 bolt
37. Safe prove separation pyro	ST+00:00:16	1/4	Truss Seq	Truss Pyro	
38. Initiate data playback transmit from storage (data Mode B)	ST+00:01:00	1/4	BVS-DSS	BVS-DS	
. . . . Prove Impact	ST+00:36:00				S/C mounted equipment shutdown
39. Initiate data Mode A (transmission)	ST+00:12:00	1/4	BVS-DSS	BVS-DS	
40. Sense balloon extraction pressure	ST+00:15:00 P-00:00:00	2	Guidance & Control	Truss	
41. Arm balloon extraction and inflation pyro	P+00:00:01	1/4	Truss Seq	Truss Pyro	
42. Fire balloon extraction pyro	P+00:00:02	2	Truss Seq	Truss Pyro	2 bolts
43. Fire balloon inflation pyro	P+00:00:05	2	Truss Seq	Truss Pyro	2 valves
44. Safe balloon extraction and inflation pyro	P+00:00:06	1/4	Truss Seq	Truss Pyro	
45. Arm BVS chute release pyro	P+00:00:26	1/4	Truss Seq	Truss Pyro	
46. Fire BVS chute release pyro	P+00:00:27	1	Truss Seq	Truss Pyro	1 cable cutter
47. Safe BVS chute release pyro	P+00:00:28	1/4	Truss Seq	Truss Pyro	
48. Arm inflation termination and inflation system release pyro	P+00:00:58	1/4	Truss Seq	Truss Pyro	
49. Fire inflation termination pyro	P+00:00:59	2	Truss Seq	Truss Pyro	1 valve
50. Fire inflation system release pyro, re-initiate BVS-DSS	P+00:01:00	1/4	Truss Seq	Truss Pyro	1 tube cutter, 2 bolts
51. Science warm up	P+00:01:01	1/4	BVS-DSS	BVS-DS & Power	
52. Science Calibration (stored data), = data Mode "C" & "D"	P+00:03:00	1/4	BVS-DSS	BVS-Science	Transmit bit clock only on TM channel
53. Arm science deployment pyro (except gas chromatograph)	P+00:03:29	1/4	BVS-DSS	BVS-Pyro	
54. Fire science deployment pyro, open M.S. solenoid valves gather initial data sample	P+00:03:30	1/4	BVS-DSS	BVS-Pyro	1 cable cutter
55. Safe science deployment pyro	P+00:03:31	1/4	BVS-DSS	BVS-Pyro	
56. Initial sample data (except gas chromatograph) initiate data Mode E	P+00:04:00	1/4	BVS-DSS	BVS-DS	Begin data transmission to earth
57. Shut down S-band X-mitter, read G.C. data into main storage	P+00:24:00	1/4	BVS-DSS	BVS-DS	
58. Initiate hourly data sample mode C-store data	P+01:00:00 & every hr thereafter	1/4	BVS-DSS	BVS-DS	
59. Initiate 8 hour data sample Mode D	P+06:00:00 & every 8 hr thereafter		BVS-DSS	BVS-DS	
60. Transmit stored data	P+08:00:00 & every 8 hr thereafter		BVS-DSS	BVS-DS	
. . . . Initiate sonde sequence	P+........ SR-00:00:00		Earth Command	BVS-DS	Repeat for sonde 2
61. Turn on sonde engineering and science and turn on BVS mounted sonde communication equipment	SR+00:00:01	1/4	BVS-DSS	Sonde Power	
62. Arm sonde release pyro	SR+00:00:59	1/4	BVS-DSS	BVS-Pyro	
63. Fire sonde release pyro and initiate data format	SR+00:01:00	1/4	BVS-DSS	BVS-Pyro	1 cable cutter
64. Safe sonde release pyro	SR+00:01:01	1/4	BVS-DSS	BVS-Pyro	
. . . . Sonde impact	Est SR+00:39:00	1			
64. Shutdown BVS mounted sonde communication equipment	SR+00:40:00	1/4	BVS-DSS	BVS Power	

[a]BVS-DSS is abbreviation for BVS data system sequencer.

[b]BVS-DS is abbreviation for BVS data system.

SUBSYSTEMS

BVS SCIENCE SUBSYSTEM

The experiment complements for the three baseline BVS mission modes have been selected to accomplish as many of the objectives as possible and were designed to be nearly identical for each mission mode. The only difference in payloads is in the replacement of the visual photometers and solar aspect angle sensors with a more elaborate light backscatter experiment on the 1973 Venus-Mercury mission since that mission is restricted to the dark side. The common BVS science payload and objectives are summarized in table 18; table 19 gives the experiment characteristics. A description of each instrument follows.

Triaxial Accelerometer

This instrument serves a dual purpose: it measures the entry vehicle deceleration and, after BVS deployment, the accelerations of the gondola caused by turbulence. This, of course, requires a dual-range instrument that measures 0 to 300 g for the entry measurements and ±1 g for the turbulence measurements. Some development may be required to produce the desired accuracy and ranges in a single instrument. A typical accelerometer triad with the required accuracy is shown in figure 27. It is felt that use of microelectronics and some redesign could reduce the weight to the 1.5 lb assumed for the baseline payload.

For the turbulence measurements the accelerometer is turned on every 6 or 8 hr and a g-switch armed to trigger sampling when the turbulence causes accelerations above a certain level, say ±0.1 g. When these levels are exceeded for a given time, the g-switch initiates sampling of the three output channels at the rate of 2 samples per second for 24 sec giving a total of 1152 bits, not including frame synchronization and time.

The accelerometer is then turned off to be armed again 6 or 8 hr later. It would be desirable to have the ability to reprogram the sampling on command from Earth since it is not possible to estimate the turbulence spectrum at this time.

TABLE 18.- BVS EXPERIMENTS AND OBJECTIVES

Experiment	Objectives
Triaxial accelerometer	Measure the turbulence and wind gusting over the BVS drift path.
Pressure, temperature sensors (density from BVS mass/volume)	Measure spatial and temporal variations of atmospheric pressure and temperature over the BVS drift path.
H_2O vapor sensor	Measure spatial and temporal variations of water vapor concentration over BVS drift path.
Light backscatter	Investigate the cloud structure and physical properties of the aerosol over the BVS drift path.
Solar aspect angle sensors	Measure solar zenith angle for interpretation of other measurements, cloud cover, and BVS position.
Visual photometers	Measure solar radiation intensity and scattering within clouds and variations over BVS drift path.
Mass spectrometer	Determine detailed atmospheric composition: concentrations of permanent gases, and concentrations and variations of trace constituents over BVS drift path.
Aerosol collector	Collect aerosol samples for analysis in gas chromatograph and biological laboratory.
Gas chromatograph	Determine detailed cloud composition, search for organic compounds, and variations over BVS drift path.
Biological laboratory	Search for life in atmosphere and clouds.
Drop sondes	Measure pressure, temperature, profiles to surface, and H_2O concentration or radiation fluxes to surface over several widely separated points.
Radar altimeter	Provide altitude reference for other measurements, determine BVS velocity over surface, average surface slopes, investigate surface reflectivity, and roughness over BVS drift path.
BVS tracking	Determine general circulation pattern at flotation altitude.
In-atmosphere occultation	Investigate neutral atmosphere and ionosphere structure and variations from measurements of telemetry signals from BVS.

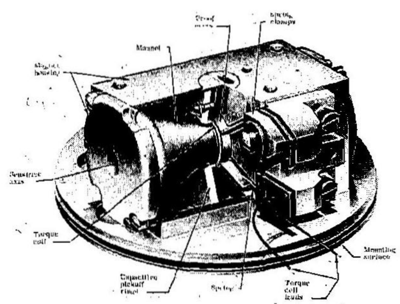
TABLE 19.- BVS EXPERIMENT CHARACTERISTICS

Experiment	Weight, lb	Power, W	Data	Development status	
				Design	Technology
Triaxial accelerometer	1.5	2.1	8 bits/axis, 2 samples/sec for 24 sec every 8 hours[c]	Current	Current
Pressure sensors (2)	1.0	2.8	8 bits/sensor, 1 sample/hr	Current	Current
Temperature sensors (2)	1.3	2.0	8 bits/sensor, 1 sample/hr	Current	Current
H_2O vapor sensor (1)	0.8	0.4	8 bits/sample, 1 sample/hr	New	New
Light backscatter[a]	[a]1.5	[a]3.9	2 detectors, 8 bits/detector, 1 sample/hr	New	Current
Solar aspect angle sensor[b] (3)	1.5	1.3	8 bits/sensor, 1 sample/hr	Current	Current
Visual photometer[b]	2.0	1.5	3 detectors, 8 bits/detector, 1 sample/hr	Current	Current
Mass spectrometer Sampling valve for M.S.	9.0	8.0	1200 bits/20 sec, every 8 hr[c]	Current	Current
Gas chromatograph Aerosol collector	7.0	5.0	1200 bits/20 minutes, every 8 hr[c]	Current	Current
Minibio lab	10.0	1 cont 2 readout	48 bits/readout, one readout/hr for 100 hr	New	New
Drop sondes (2)	5.0 each			New	New
Radar altimeter	12.5	20.0	8 bits (altitude), 3x4 bits (velocity), 4 bits (reflectivity), total 24 bits per sample, 1 sample/hr	New	Current
Total weight	58.1				

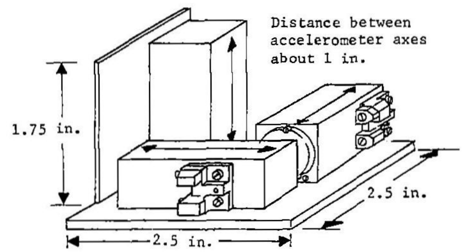
[a] Weight and power increased to 5 lb, 6 W for 1973 Venus-Mercury configuration.

[b] Removed from 1973 Venus-Mercury mission configuration.

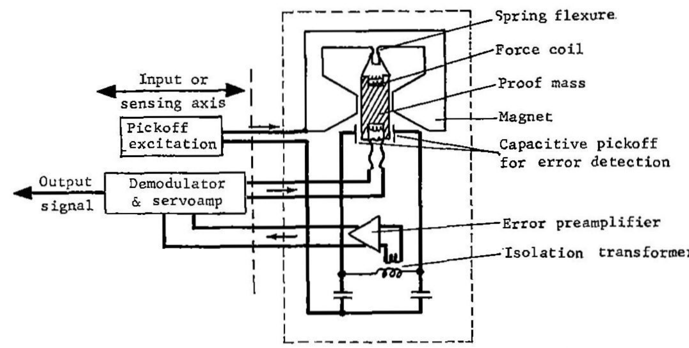
[c] Once every 6 hr for 1973 BVS/orbiter mission configuration.



Bell Aerosystems model VIIB accelerometer



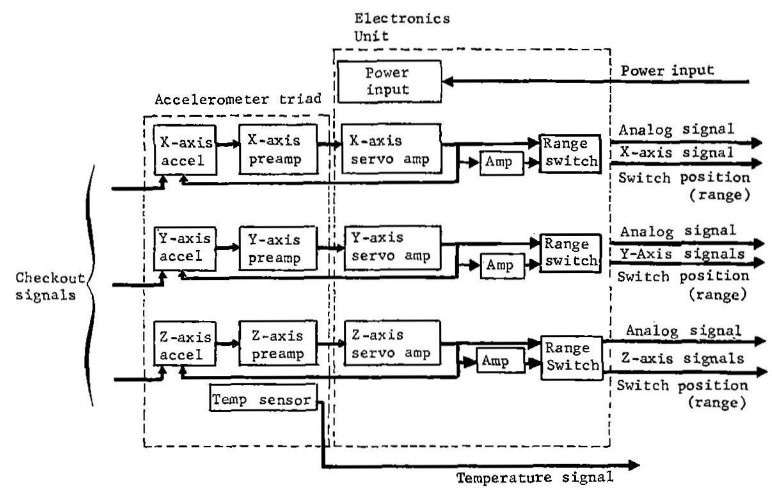
Accelerometer triad



Single-axis accelerometer with servo loop

Characteristics of accelerometer triad and associated electronics

	Accelerometer Triad	Associated Electronics
Weight	20 oz	16 oz
Power	5 w	10 w
Size	2.5x2.5x1.75 in.	9 cu in.
Acceleration ranges		
Parallel to roll axis	0 to 320 g, 0 to 25 g 0 to 2 g	
Normal to roll axis	0 to 20 g, 0 to 1 g	
Accuracy	±0.1% fs	
Data rate, capsule entry	24 bits/sample, 20 samples/sec	
Data rate, BVS	24 bits/sample 2 samples/sec	
Status	Basically developed; sterilizable unit can be available by 1969	
Mounting	On roll axis near cg of capsule	



Block diagram of triaxial accelerometer system

Figure 27.- Typical Triaxial Accelerometer

Pressure Sensors

Two pressure sensors with overlapping ranges (e.g., 400 to 1200 mb and 500 to 700 mb) are mounted inside the BVS gondola, which is vented to the atmosphere. The low range provides for the measurement of small pressure variations while the wider range provides both redundancy and a capability for obtaining a limited pressure profile during the final descent of the BVS. As in the previous study (ref. 4), the characteristics of a nonexistent "typical" pressure sensor have been assumed since there are many such devices presently available.

Temperature Sensors

There are two temperature sensors on the BVS; one is mounted on a short boom attached to the gondola while the other is deployed on a thin cable 29 ft below the balloon center to reach conditions unaffected by the presence of the BVS. The sensors are shielded platinum resistance elements. Each has a range from 200 to 400°K. The instruments are described in more detail in reference 1.

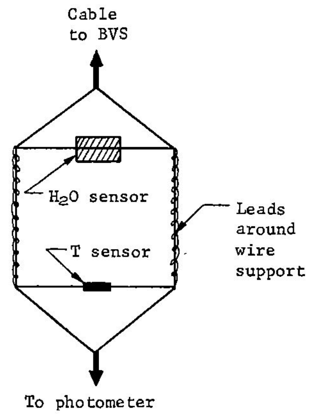
Density

The atmospheric density can be determined since the BVS mass and volume are known. Since the BVS floats at a constant density level, changes will occur only when ballast is dropped (i.e., drop sondes). Alternatively, the pressure, temperature, and mean molecular weight can be used to verify the BVS mass/volume.

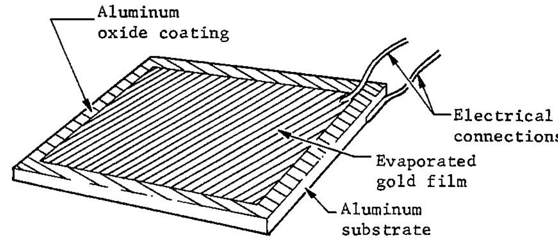
Water Vapor Sensor

At present, no completely satisfactory (unambiguous) method exists for determining the water vapor content of an unknown (other than CO_2, etc.) atmosphere. An aluminum oxide sensor has been assumed for this tudy, but unknown trace constituents (or $HC\ell$) may have some effect on the measurements. The question should be examined experimentally before the final choice of a sensor is made.

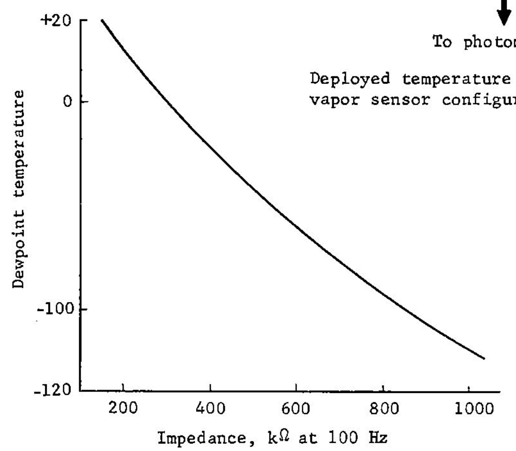
In the baseline configuration, the H_2O sensor is deployed with the temperature sensor 29 ft below the balloon center to minimize effects of the BVS. Figure 28 shows the sensor configuration and mounting. Such a sensor is being developed at JPL. Sampling is once per hour but it would be desirable to have the ability to change the sampling rate in flight to monitor more rapid variations.



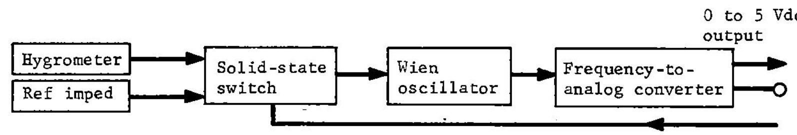
Deployed temperature sensor and H_2O vapor sensor configuration



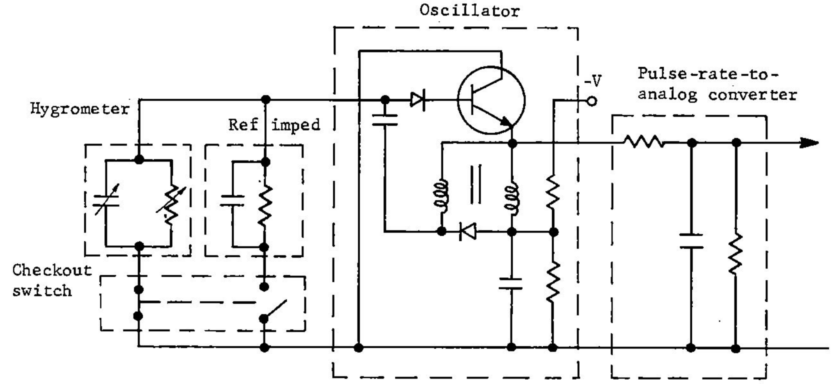
Construction of aluminum oxide sensor



Typical characteristic of sensor
Impedance vs dewpoint temperature



Block diagram for hygrometer instrument



Simplified electronic circuit diagram for hygrometer

Hygrometer characteristics

Weight of support with sensor: about 0.3 lb
Weight of associated electronics: 0.5 lb
Power: 0.4 W
Size of sensor: 0.2x0.6x0.003 in. is typical
Volume of electronics: 2 cu in.
Range of sensor dewpoint temperature: -150° to +10°C
Temperature range of electronics: -16° to +80°C
Data rate: 8 bits/sample, 1 sample/hr
Time constant: about 1 sec claimed for 63% change

Figure 28.- Humidity Sensor

Light Backscatter Experiment

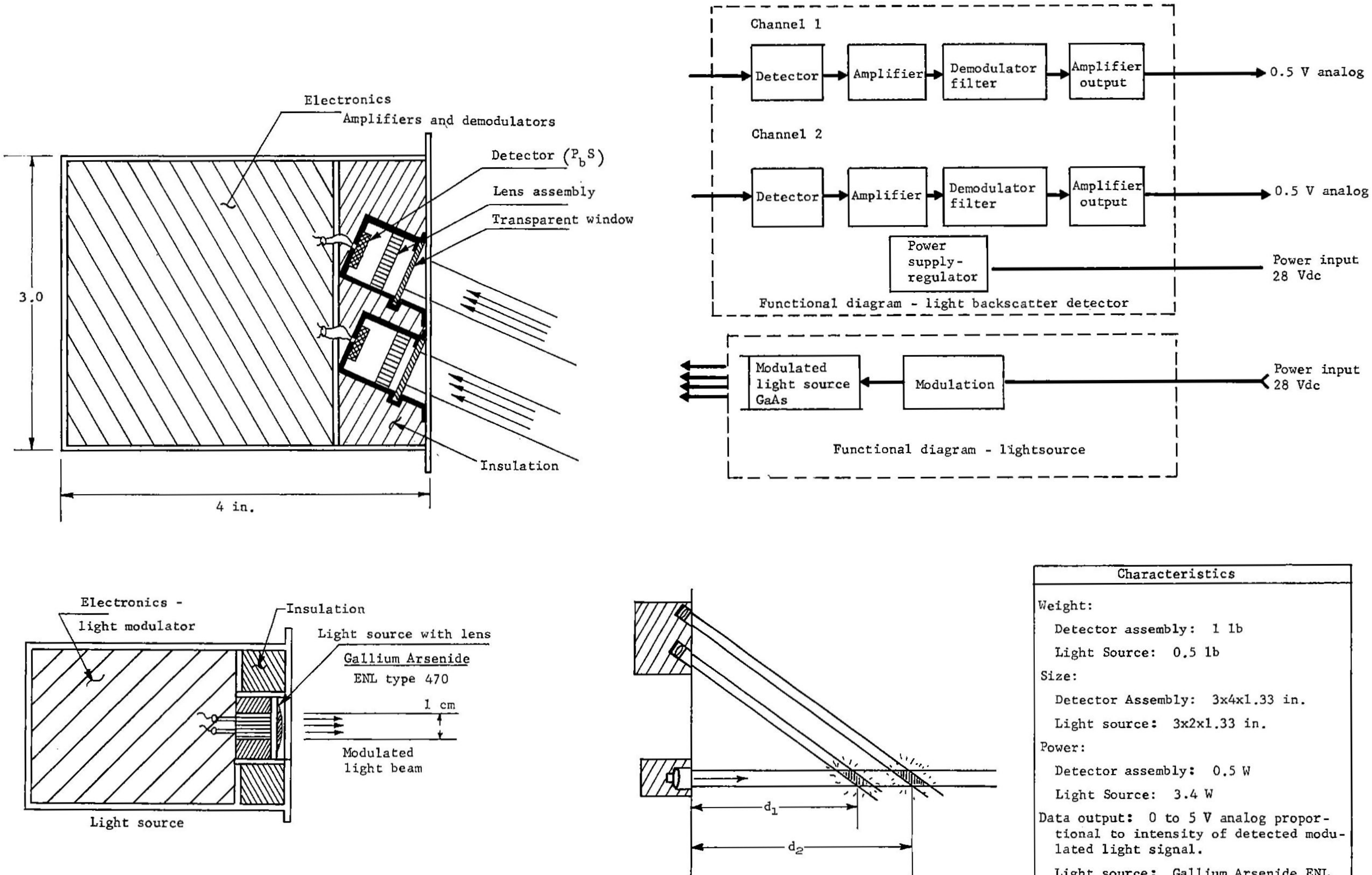
The objective of this experiment is to monitor the presence of clouds in the immediate vicinity of the BVS and to investigate the nature of the aerosol (e.g., particle sizes, size distribution, concentration, refractive indexes, etc.). The basic concept is to observe the intensity and/or polarization of the light backscattered from the cloud particles at several angles and wavelengths and to determine from this information the nature of the particles. The extent to which these physical properties can be determined depends on the degree of sophistication in the instrumentation.

For the purposes of this study, a rather simple experiment has been assumed although there are many other more sophisticated options possible. The experiment used for the baseline payload is illustrated in figure 29. The light source is a gallium arsenide (GaAs) diode electronically modulated to help discriminate the backscattered light from the background light and variations due to changes in the solar light intensity. Peak output is 6×10^{-3} W/sr at 9100 Å (peak) and 300°K.

The two detectors, with suitable filters (perhaps polarizing) are physically separated from the source to accept light backscattered from two volume elements as shown in figure 29. The detector signals are electronically demodulated, averaged over a short time, and digitized to an 8-bit sample. Sampling is once per hour although it would be desirable to increase this rate on command from Earth.

Solar Aspect Angle Sensors

These instruments are used to help determine the BVS position and interpret the visual photometer measurements. Figure 30 illustrates the functional operation of the Adcole Digital Solar Aspect Sensor. The actual instrument consists of two such sensors at right angles to one another as shown. Three sensors are located 120° apart around the periphery of the BVS gondola. The balloon is transparent so the sun can be viewed through it; however, it acts as a spherical lens and its effects on the aspect angle must be calibrated. Another solution would be to mount the sensor on the balloon surface itself.



Characteristics

Weight:
- Detector assembly: 1 lb
- Light Source: 0.5 lb

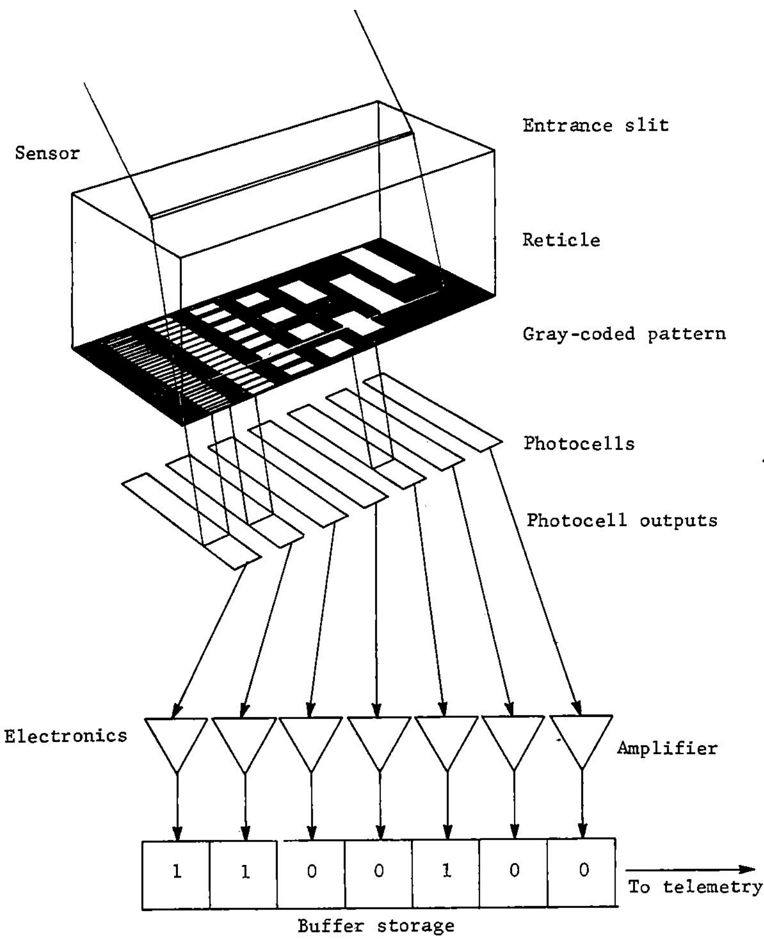
Size:
- Detector Assembly: 3x4x1.33 in.
- Light source: 3x2x1.33 in.

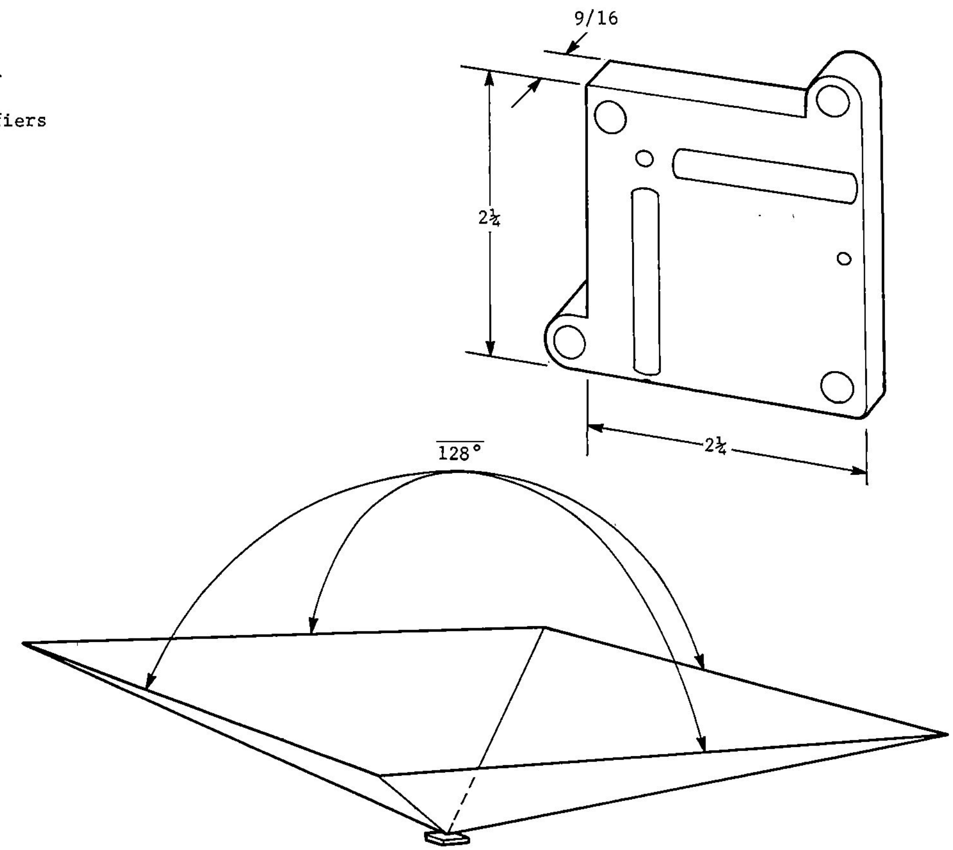
Power:
- Detector assembly: 0.5 W
- Light Source: 3.4 W

Data output: 0 to 5 V analog proportional to intensity of detected modulated light signal.
- Light source: Gallium Arsenide ENL type 470
- Peak wavelength: 9100Å

Figure 29.- Light Backscatter Instrument

Functional operation (simplified) of Adcole system showing sensor, comprised of reticle and photocells; and electronics, comprised of amplifiers and buffer storage





Field of view of sensor

Sensor weight (3)	0.75 lb
Electronics weight	0.75 lb
Power	
Field of view	±64°
Angular resolution	1°
Size (sensor)	2.25x2.25x9/16 in.

Figure 30.- Solar Aspect Sensor

Visual Photometers

The three-channel visual photometer is deployed on the cable with the temperature and H_2O sensors and about 2 ft below them (~31 ft below balloon center). The balloon subtends an angle of 32.4° at this distance so a direct view of the sun is obtained if the zenith angle is greater than 16.2°. Figure 31 summarizes the characteristics of the visual photometer.

Mass Spectrometer

The mass spectrometer determines the concentration of gases with atomic mass numbers between 1 and 90 amu. This range includes most of the common atmospheric gases (e.g., O_2, CO_2, N_2, A) and their dissociation products. There are several problems associated with the mechanization of this experiment. First, the sampling ports must be located to sample atmosphere that is uncontaminated or otherwise affected by the presence of the BVS. One solution is to deploy a long tube for sampling. However, the tube itself could contaminate the sample or react with it. At this time there is no proved solution to the problem and in the baseline configuration the sample is introduced through a short tube through the gondola, providing as short a path as possible to the ion source.

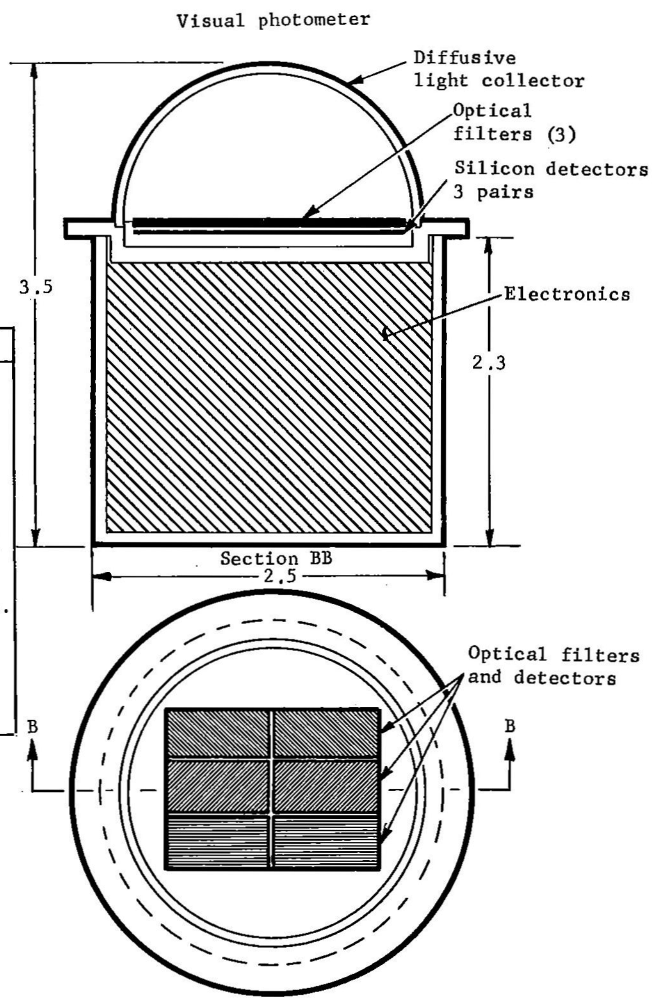
The other major problem concerns the maintenance of a vacuum in the device over long periods of time. Judicious programing of the sampling times is required as well as the development of a vacuum-tight sampling valve.

Figures 32 and 33 illustrate the characteristics of two types of mass spectrometers. Both are suitable for this application. The instruments are being developed at JPL and Goddard SFC.

Gas Chromatograph

This instrument is used to determine the cloud composition, and to resolve atmospheric gases with the same mass numbers (e.g., N_2 and CO_2). A small (~2 lb) collection system (fan and filter) is used to gather a sample of the aerosol and deliver it to an oven for pyrolysis. Three columns should be sufficient -- one for separation of atmospheric gases and two for the cloud particle analysis. As with the mass spectrometer, sample contamination or modification by the sampling system presents a problem.

Characteristics of visual photometer	
Weight:	2 lb
Volume:	28 cu in.
Power:	1.5 W at 28 Vdc
Data:	3 data channels - one for each spectrum - digitized to 8 bits in DAS, sampled at 10/sec rate. 1 temperature data channel digitized to 8 bits.
Silicon detector:	Spectral response; 0.4 to 1.1 μ.
Optical filters:	Band no. 1, 0.4 to 0.5 μ Band no. 2, 0.5 to 0.6 μ Band no. 3, 0.6 to 0.7 μ
Filter transmission:	80% efficiency



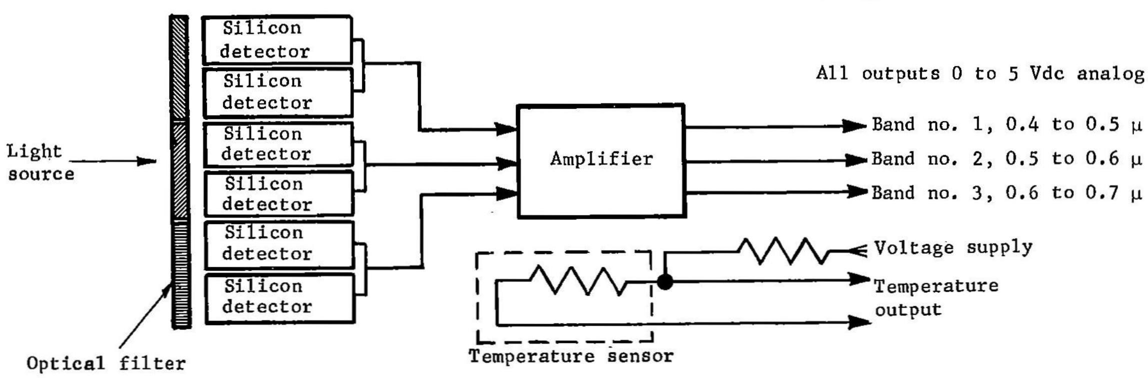
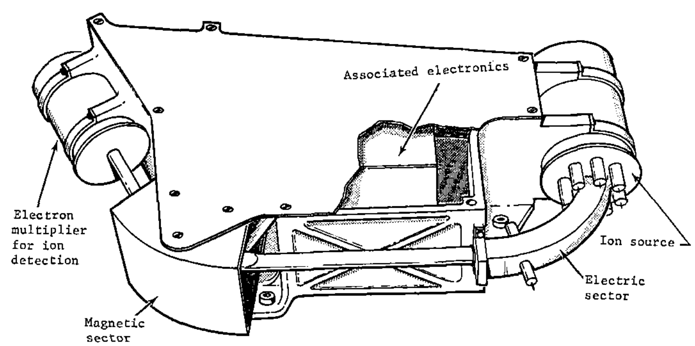
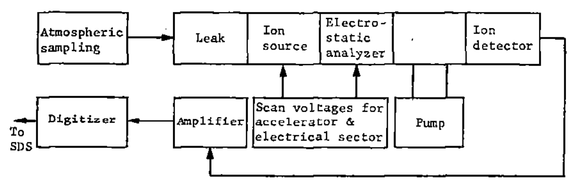


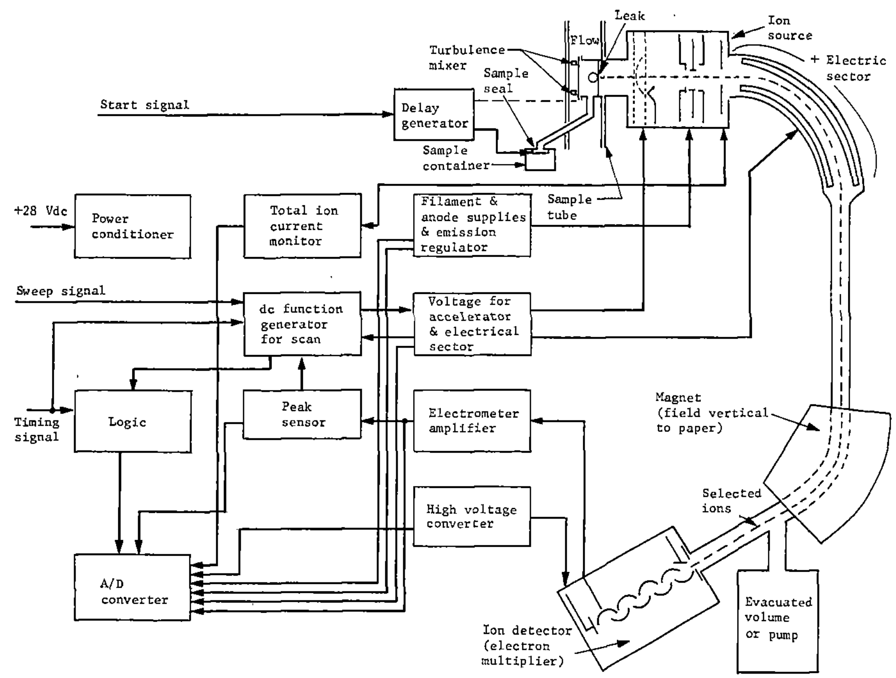
Figure 31.- Visual Photometer



Configuration of a double focusing mass spectrometer Package



Basic block diagram of double focusing mass spectrometer



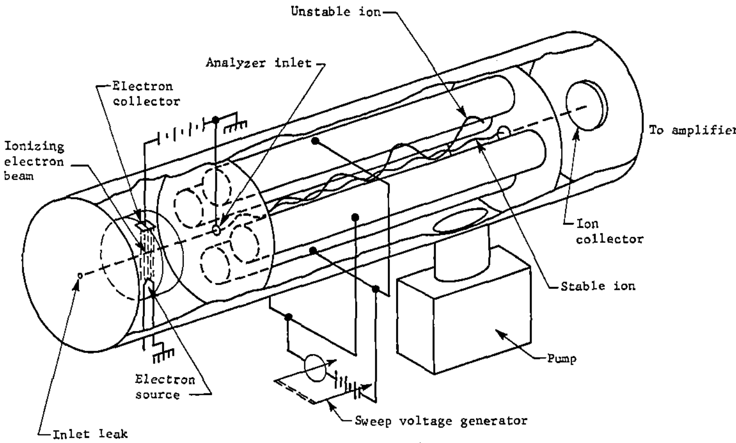
Functional schematic for double focusing mass spectrometer

Characteristics of double focusing mass spectrometer

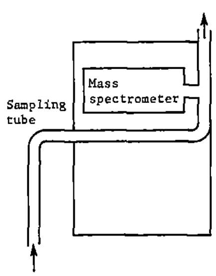
Weight, total: 7 lb
Power: 5 w
Size: about 13x8.6x2.7 in.
Mass range: about m/e 10 to m/e 80
Scan time: about 2 sec
Resultuion: $M/\Delta M$ = 150 at 100 amu
Detector dynamic range: 10^4

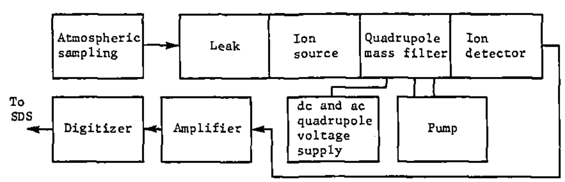
Data output:	About 70-7 bit words for mass spectrometer scan (with log. electrometer ampl).
	About 5-7 bit words per scan for engineering measurements.
Data readout rate:	1 to 2 mass spectrometer scans every 8 sec.
Mode of operation:	Scan of mass number range with peak searching and pressure measurement.

Figure 32.- Double Focusing Mass Spectrometer



Basic configuration of quadrupole mass spectrometer



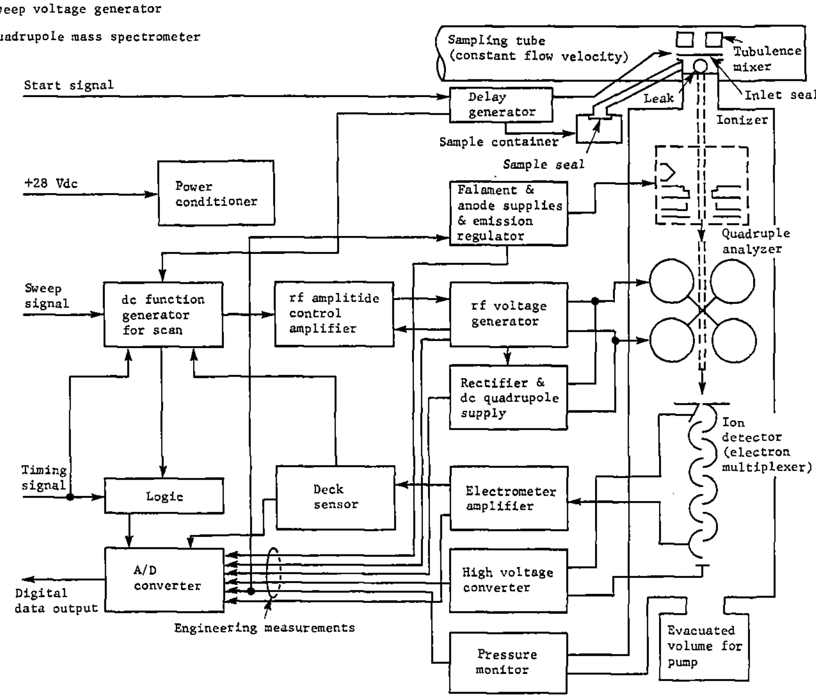


Basic block diagram of quadrupole mass spectrometer

Characteristics of quadrupole mass spectrometer

Weight total:	8 lb
Power:	10 w
Size:	8.6x14x2.8 in. plus gas entrance configuration
Mass range:	m/e 10 to m/e 60
Scan time:	2 sec or more
Resolution:	M/ΔM - 50 at 30 AMU
Accuracy:	10% on 1% constituents 5% on 80% constituents

Data output assumed:	50-7 bit words for mass spectrometer scan 6-7 bit words per scan for engineering measurements
Data readout rate:	2 mass spectrometer scans every 8 seconds
Mode of operation:	Scan of mass number range with peak searching and "ion gage" pressure measurement
Detector dynamic range:	10^4



Functional schematic for quadrupole mass spectrometer

Figure 33.- Quadrupole Mass Spectrometer

The weight and power for the instrument used in the baseline configuration imply a rather limited analysis capability -- that is, the instrument will probably be able to detect only specified constituents rather than search for and identify unknown constituents. The increased sophistication to identify unknown compounds could be obtained with an additional 10 lb -- i.e., by using the Bio Lab weight allocation to increase the capability of the gas chromatograph.

Bio Lab

In lieu of defining a specific life detection experiment, a 10-lb weight allocation was reserved for this type of experiment and a typical data rate assumed. Typical life detection experiments that can be accommodated are described in reference 4. One possible option for this 10-lb allocation would be to include a more sophisticated gas chromatograph system that could search for organic compounds as well as perform more detailed cloud composition analysis.

The aerosol collector used for the gas chromatograph also collects and delivers samples to the Bio Lab.

Radar Altimeter

The radar altimeter provides an altitude reference for the other measurements. It also measures the BVS velocity over the surface and surface reflectivity. It is a pulsed, X-band, scanned beam, phased array radar. The transmitter consists of an 8x8 phased array of Radio Frequency Building Blocks (RFBB), each of which contain frequency multipliers, phase shifters, power amplifiers, receiver mixers, IF amplifiers, and solid state switching elements in a sealed package 2.85x0.9x0.3 in. The auxiliary circuitry (power supply, data processing, timing, and RF source) is packaged separately and manifolded to the RFBB array. Table 20 lists the characteristics of the instrument. The instrument has been developed by Texas Instruments under Contracts AF33(615)-1993 and NAS1-5954. A detailed description of the RFBBs is given in NASA CR-66383.

TABLE 20.- RADAR ALTIMETER SPECIFICATIONS

Size, cu in.	450
Weight, lb	12.5
Power	20 W at 28 Vdc
Peak RF power, W	64
Scan angle	20° off vertical
Range, km	60 to 70
Beam width	10° (steerable)
Accuracy, %	2

Drop Sondes

The two drop sondes assumed for the baseline configurations are based on those discussed in reference 4. Each 5-lb sonde carries a temperature sensor (200 to 800°K), two pressure sensors (0 to 15 and 0 to 150 atm) and either a water vapor detector or a visible light photometer. Figure 34 is a schematic illustration of a small sonde. Further details are given in reference 4.

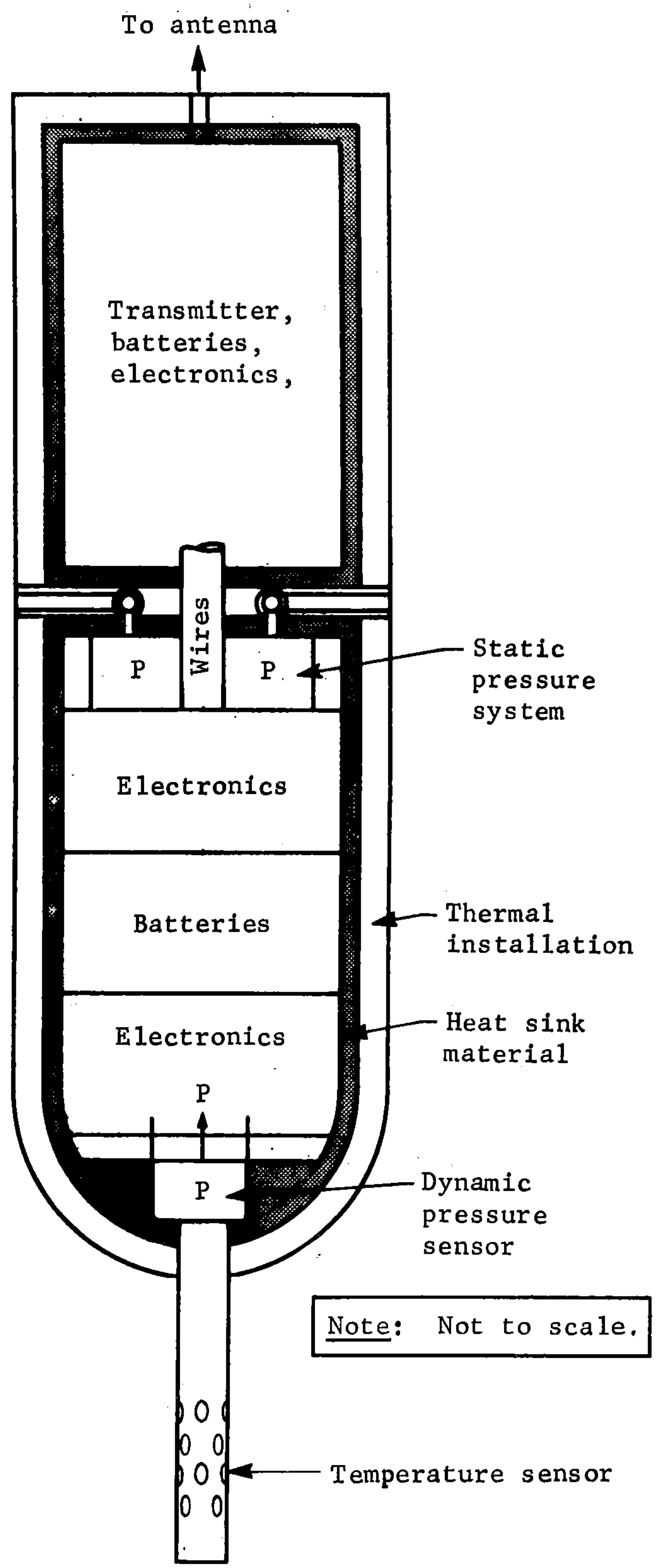


Figure 34.- Schematic of Small Sonde

BVS TELECOMMUNICATIONS

Telecommunications for three types of missions have been considered -- Venus orbital, Venus flyby, and Venus/Mercury flyby. A baseline configuration and options with tradeoff considerations are described for each of the missions.

For the orbital mission, relay communications between the BVS/entry vehicle and the orbiter are used for telemetry, command, and Doppler measurements. Noncoherent FSK is used for both telemetry and command.

A phase lock loop is used in the BVS to track a continuous mark frequency transmitted by the spacecraft (following command transmissions) to obtain one way doppler measurements*. These data are stored in the station along with science and engineering data, which are sampled on a scheduled basis during intervals between communications contact with the spacecraft.

A fixed telemetry bit rate of 240 bps is attained using a 20 W 390 MHz transmitter in the BVS for both entry and BVS floatation operations. The BVS transmitter handles entry data for the capsule as well as the BVS during preentry and entry phases of the mission.

The spacecraft to-BVS command system uses tone sequential modulation for six commands that are generated by varying the frequency shift key rate of the 20 W 410 MHz spacecraft transmitter.

Both entry from approach and entry from orbit were considered. The relay link performance for either approach during entry is comparable, each with a view time for entry and deploy operations of at least 30 minutes.

During orbital operations, power to the BVS transmitter is applied as a result of detecting the spacecraft beacon and a 10-minute on-time is allowed for transmission of data to the spacecraft. Drift of the station (from an initial position 65° from periapsis) in any direction over a central angle of 30 to 40° continues to provide a 10-minute view time; however, a 5-minute view time is adequate for transfer of stored data on a nonrepeat basis.

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*An optional configuration to retain the noncoherent FSK but at the same time obtain ranging and two-way doppler measurements by the orbiter for BVS position determination is presented in the appendixes.

For the Venus flyby mission, entry data are via relay link as in the orbiter mission. Postdeploy communications contacts are direct to Earth at a bit rate of 30 bps using an S-band 20 W transmitter and single subcarrier PCM/PSK/PM modulation (no separate sync subcarrier is used). Commands to the BVS are direct from Earth using the standard 1 bps Mariner-type command system. Contacts with Earth are scheduled on a once per 8-hr basis with provisions for contact on any hour for command purposes. Transmitter on-time has been limited to 20 minutes for a given contact with stored electrical energy sufficient for a 50-hr minimum mission duration.

For the Venus/Mercury flyby mission, both entry and postdeploy operations require a direct link to Earth. The command system is identical to that for the Venus flyby mission. The telemetry link to Earth operates at a bit rate of 120 bps.

During the high g entry (above 2g) and during blackout the entry data are stored in the BVS. Real-time telemetry appears feasible below 2 g acceleration, but predetection recording at the Deep Space Net Station is required for backup processing (in case of real-time acquisition problems) and the processing of signals transmitted during the high g portion of entry. The postdeploy station operations and transmission duty cycle is the same as for the Venus/flyby mission.

The conclusions of this design effort are listed in table 21.

TABLE 21.- DESIGN EFFORT CONCLUSIONS

1. The telecommunications requirements can be met for all missions studied with state-of-the-art concepts.
2. Relay links provide better data rates for entry phases (for the same transmitter power) than do direct links when entry is in the spacecraft orbit plane and the range to Earth is approximately 0.3 A.U. or beyond.
3. Communications geometry and performance margins for both entry from approach and entry from orbit are comparable.
4. Relay links between a BVS and a flyby spacecraft following planet encounter are not practical in most cases. Direct links to Earth are therefore required.
5. Use of noncoherent FSK modulation with its inherently lower data rate (for relay communications) is preferred for relay links as compared to coherent PSK with its continuous frequency search and acquisition demands.
6. Block coding could be used on BVS to DSN telemetry links to improve data rate capability by a factor of approximately 1.7. However, for the present, this approach is held in reserve as a growth potential since modification of the DSN is involved.
7. The planned standard Mariner-type command and multiple mission telemetry hardware facilities of the NASA DSN are adequate to support the BVS missions. However, if two-way coherent communications lock up with the DSN is used simultaneously for both the spacecraft and the BVS, it appears that two stations must support the encounter and entry phases simultaneously.

Telecommunications for the Orbital Mission

Radio subsystem - The requirements for the BVS/entry vehicle radio subsystem are shown in table 22. Although the subsystem is located entirely in the BVS, which is separated from the aeroshell after entry, it fulfills the requirements for transmitting all BVS/entry vehicle telemetry data associated with separation from the spacecraft, ΔV burn, and entry, as well as fulfilling the communications requirements for the BVS during subsequent contacts with the orbiter.

TABLE 22.- BVS RADIO SUBSYSTEM REQUIREMENTS

1. Provide a data transmission link to the supporting spacecraft during separation, deorbit, entry, and once per orbit during postdeploy opportunities.
2. Receive and detect beacon/command signals radiated from the spacecraft.
3. Provide a doppler output from the BVS receiver that is proportional to the difference between the frequency of the incoming signal and the receiver local oscillator.
4. Provide an antenna that has a radiation pattern giving essentially a hemispherical coverage.
5. Use the antenna for both receiving and transmitting.

The radio subsystem consists of an antenna, diplexer, receiver, and transmitter as shown in figure 35.

The antenna and diplexer are common to both the receive and transmit circuits. A crossed-slot, cavity-backed antenna having hemispherical coverage and an on-axis gain of 5 dB is used. Antenna characteristics are shown in figure 36. Transmission before separation of the BVS from the aeroshell is through a radome in the afterbody of the aeroshell.

The receiver is used to receive spacecraft command/beacon radio tranmissions that signal the BVS to begin radio transmissions and specify operating modes. The receiver is designed for frequency shift key (FSK) tone modulation. Mark and space predetection filters are followed by square law detectors and a summing circuit. The keying rate is varied to provide a tone command capability. A doppler output is also provided that can be operative when an unmodulated carrier at the mark frequency is transmitted from the spacecraft. Other receiver characteristics are shown in table 23.

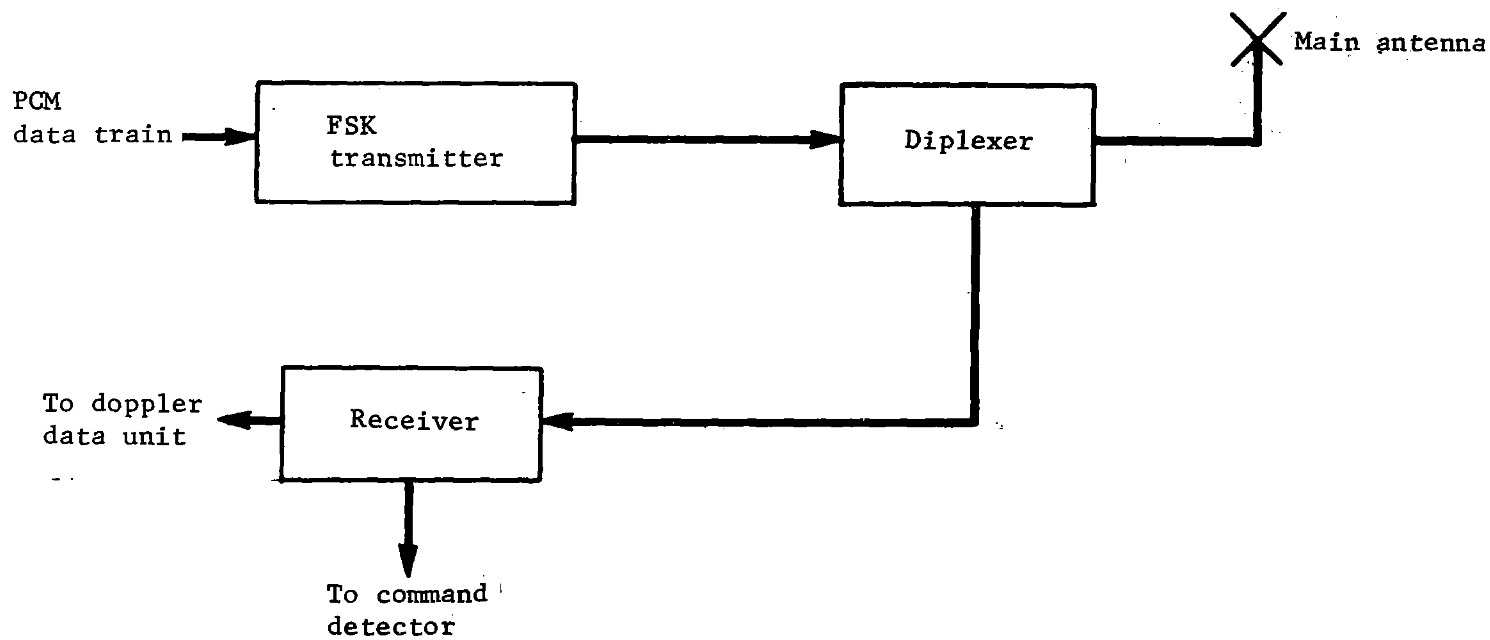
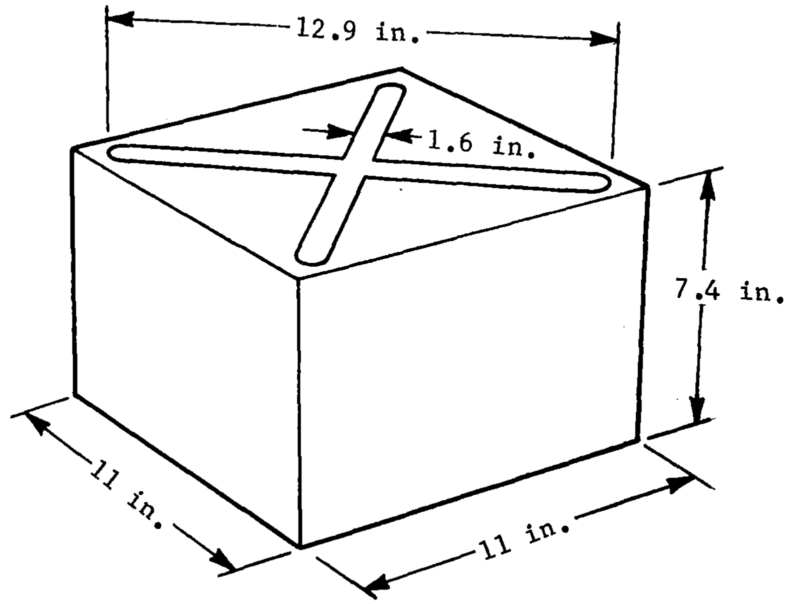


Figure 35.- BVS Radio Subsystem Block Diagram



Type	Crossed-slot, cavity-backed, foam-filled
On axis gain	5.0 dB
Half Power Beamwidth	120°
Center Frequency	400 MHz
Bandwidth	40 MHz
Dimensions	Approximately as shown
Weight	Approximately 5 lb

Figure 36.- BVS Antenna Characteristics

TABLE 23.- BVS RECEIVER CHARACTERISTICS

Type	Double superheterodyne FSK with space and mark predetection filters followed by square law detectors and summer.
Frequency	Mark channel - 409.695 MHz Space channel - 409.305 MHz
Local oscillator stability	1 part 10^6 short term, 1 part 10^7 long term
Noise figure	3 dB
Image rejection	80 dB
Spurious response rejection	60 dB
Dynamic range	60 dB
Power input	1.5 W
Voltage	24 to 36 V
Weight	3 lb
Volume	60 cu in. maximum

The transmitter is frequency shift keyed with a split phase NRZ data train. It operates on a mark frequency of 390.195 MHz at a bit rate of 240 bps and a power output of either 0.5 W or 20 W, depending on the control signal. The 0.5 W output is used during checkout, separation of the capsule from the spacecraft, and during ΔV burn following separation from the spacecraft.

The "mark" and "space" frequencies are generated by two crystal controlled oscillators. Selection of the frequency is made by a solid-state switch in response to the modulating data train. Other characteristics of the transmitter are shown in table 24.

TABLE 24.- BVS TRANSMITTER CHARACTERISTICS

Type	Crystal controlled dual oscillator frequency shift key modulation.
Power output	20 W minimum or 0.5 W (low power mode)
Frequency	Mark: 390.195 MHz Space: 389.805 MHz
Frequency stability . . .	1 part 10^5 long term, 1 part 10^6 short term
Modulation input	Split phase NRZ at 240 bps
Modulator	Solid state switch
Weight	4 lb maximum
Size	96 cu in. maximum
Voltage input	24 to 36 V

The BVS radio subsystem equipment is listed in table 25. Development status and estimated weight and power are also shown.

TABLE 25.- BVS RADIO SUBSYSTEM POWER AND WEIGHT SUMMARY

Unit	Weight, lb	Power, W
Transmitter, 390 MHz, 20 W	4.0	67
Main antenna	5.0	----
Diplexer	0.6	----
Command/beacon receiver	3.0	1.5
	12.6	
Development status: All equipment requirements are well within the state of the art, but design to unique specifications is required. For example, the environment is a unique requirement of the design specification.		

BVS-to-spacecraft telemetry performance (entry from orbit): The BVS-to-spacecraft relay telemetry link performance margin and associated geometry for the entry and station deployment phases of the mission are described in this subsection. The time period covered is from 10 minutes before entry through encounter of a 20° above the horizon elevation mask as the spacecraft speeds toward the horizon. Included in the analysis are effects of multipath, antenna gain pattern, the bounding atmosphere models, and targeting capability.

Performance of command and telemetry links during postdeploy phases of the mission are described in the two following sections of this report.

An entry flight path angle γ_E of -30°, targeting parameter β of 65°, and a spacecraft lead angle λ of -55° at the time of capsule entry were selected for the baseline deorbit condition. These parameters are defined pictorially in figure 37. Variations in BVS antenna aspect angle α_p, spacecraft antenna aspect angle α_B, and communications range ρ_c as a function of time are shown for the baseline conditions in figures 38 and 39.

The selected geometry results in near minimum deorbit velocity, short coast time, and no requirement for reorientation of the capsule after separation from the spacecraft. More than 30 minutes minimum communications time after start of entry is provided while maximum range is limited to 21 000 km, and the minimum elevation angle is 20° above the horizon.

Maximum communications range and near-minimum spacecraft antenna aspect angle occur near entry and exhibit little change until after BVS/aeroshell separation. The BVS antenna aspect angle experiences its maximum change during this period, resulting in increased BVS antenna pointing loss. The increased pointing loss coupled with near-maximum communications range results in a worst-case system performance margin occurring near BVS/aeroshell separation. A representation of BVS antenna gain versus aspect angle is given in figure 40.

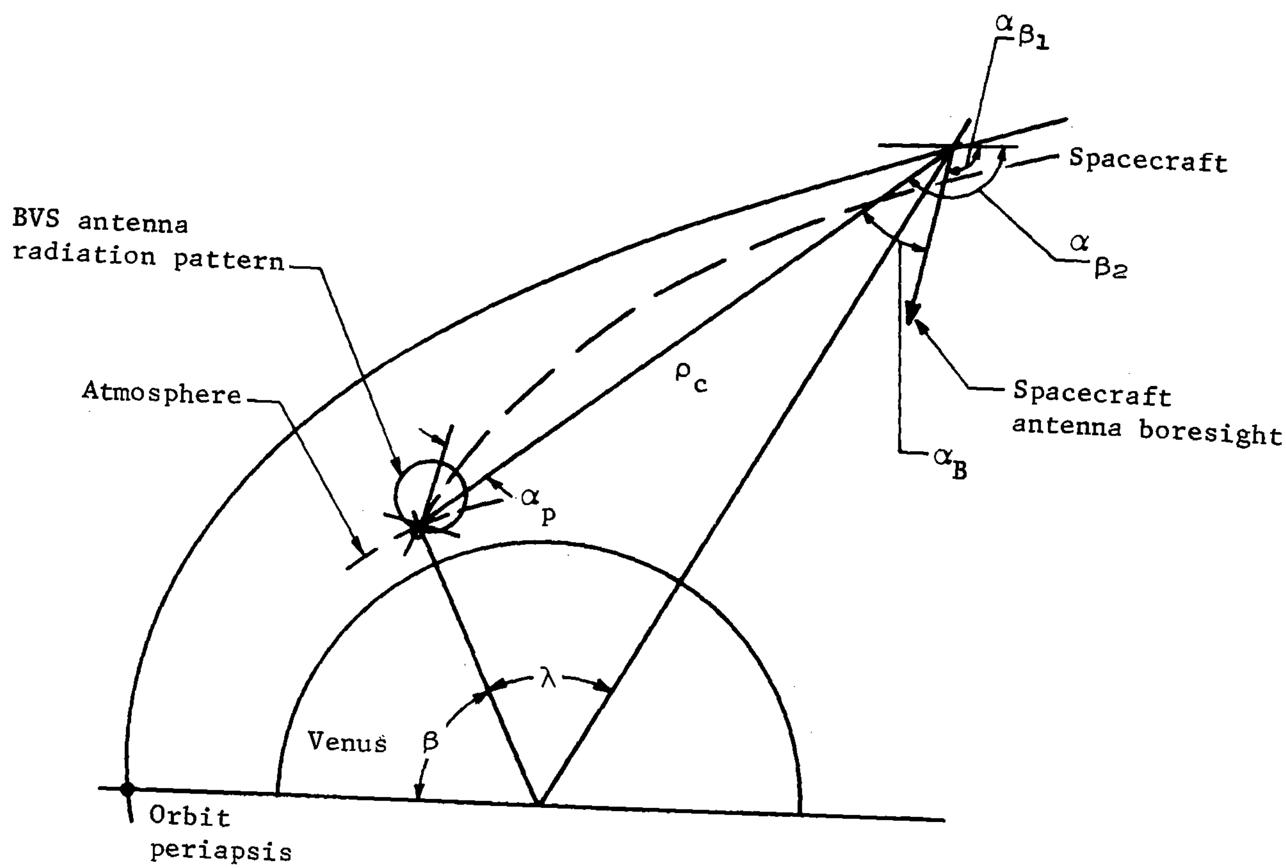


Figure 37.- Entry Geometry Convention

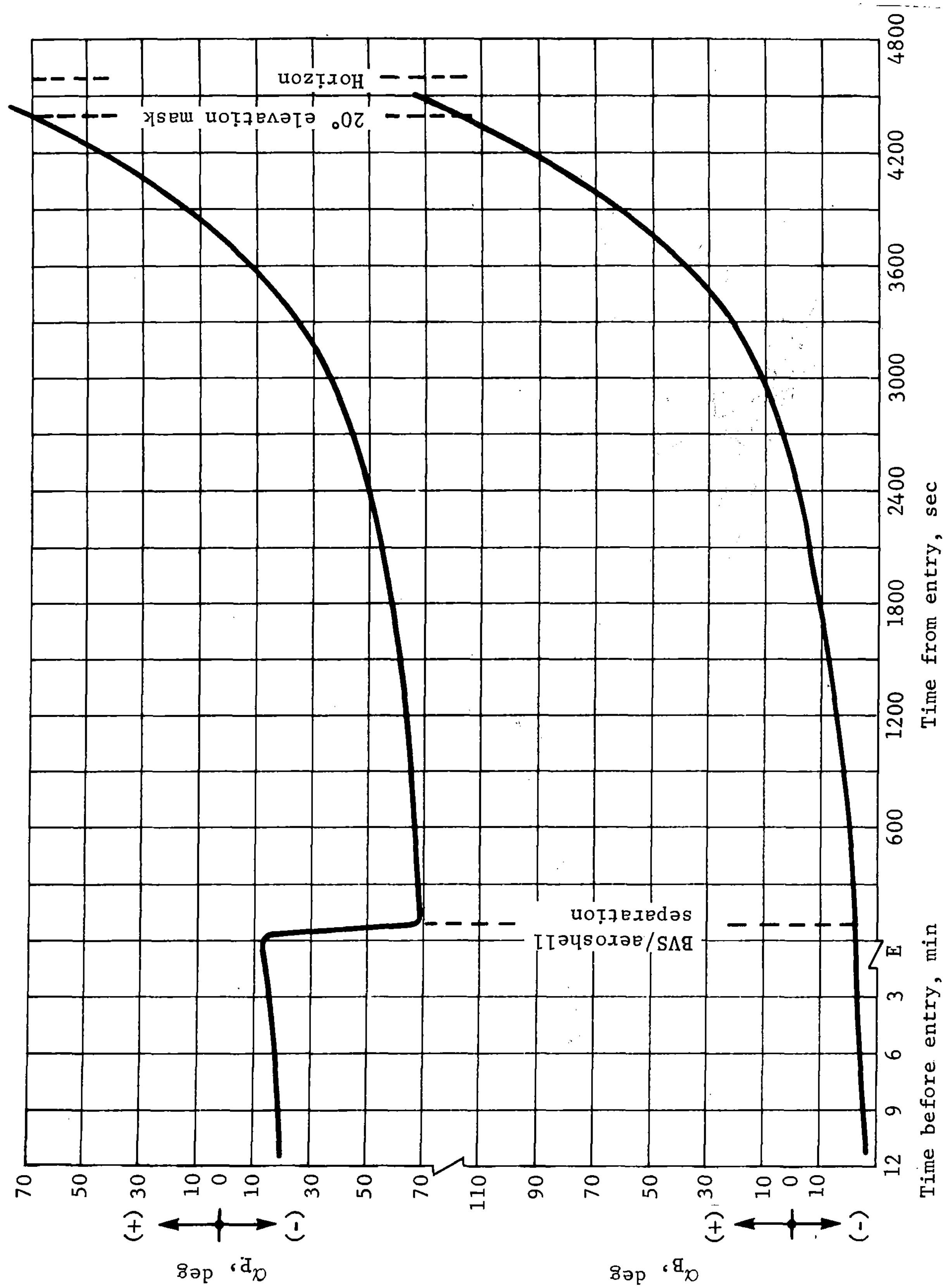


Figure 38.- Entry from Orbit Geometry (Antenna Aspect Angles)

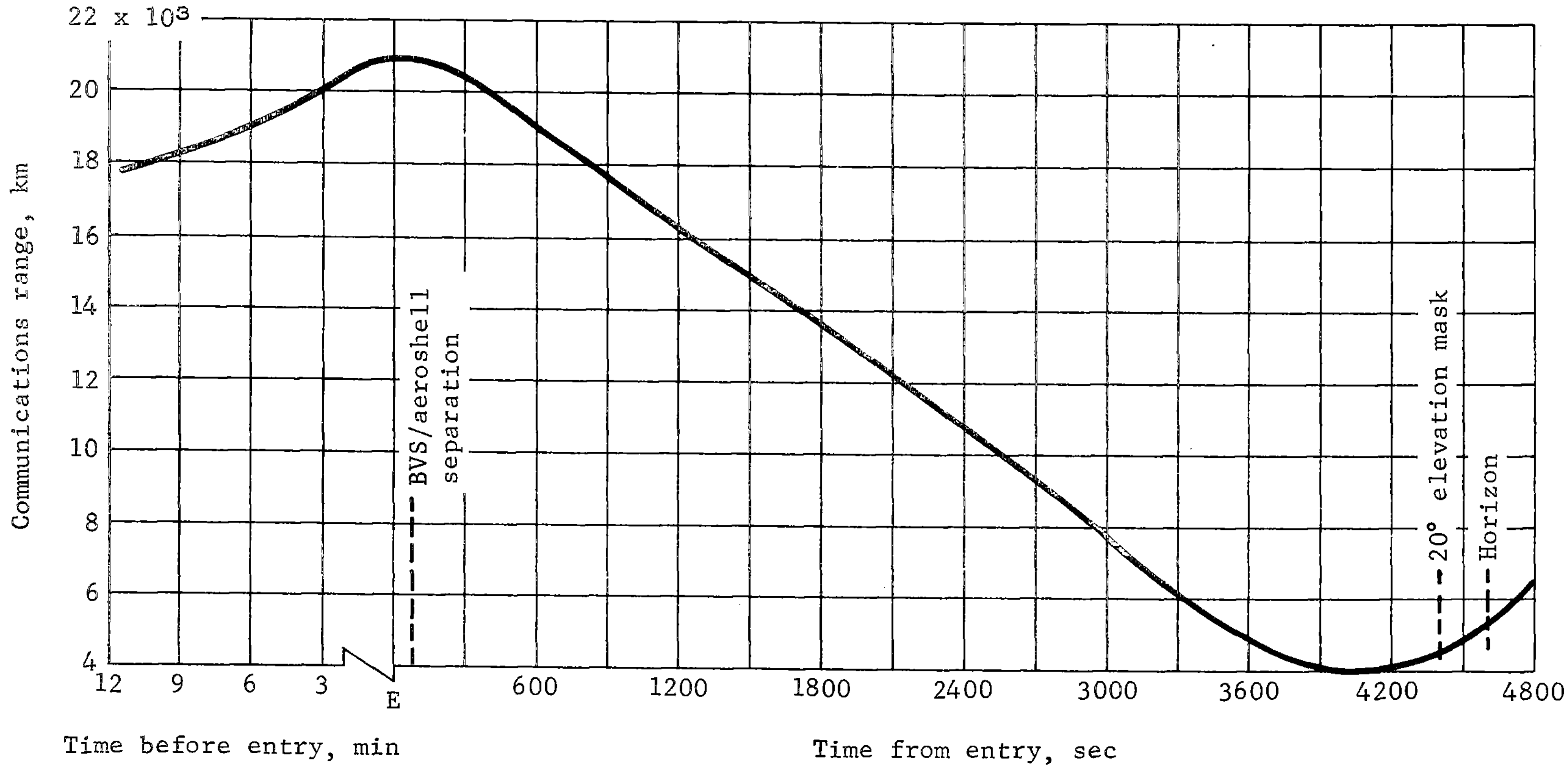


Figure 39.- Entry from Orbit Geometry (Communications Range)

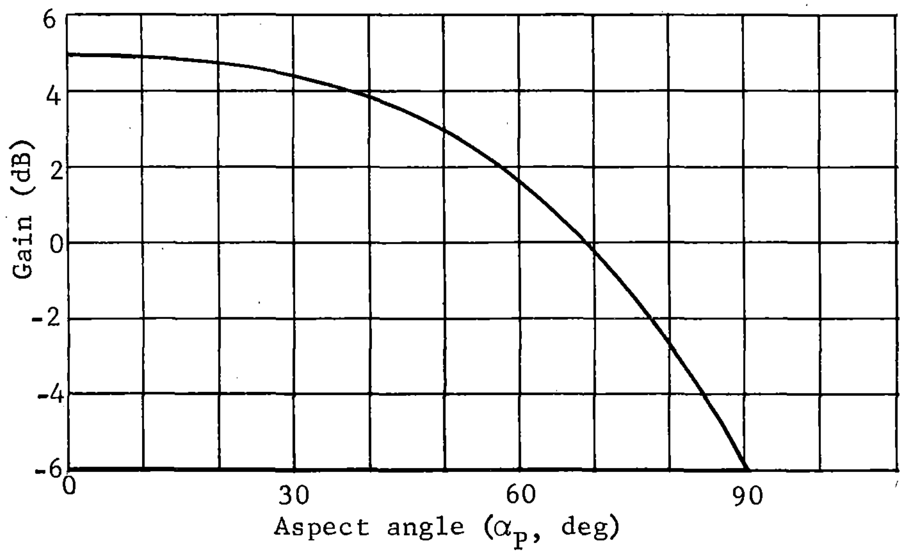


Figure 40.- BVS Antenna Gain Patterm

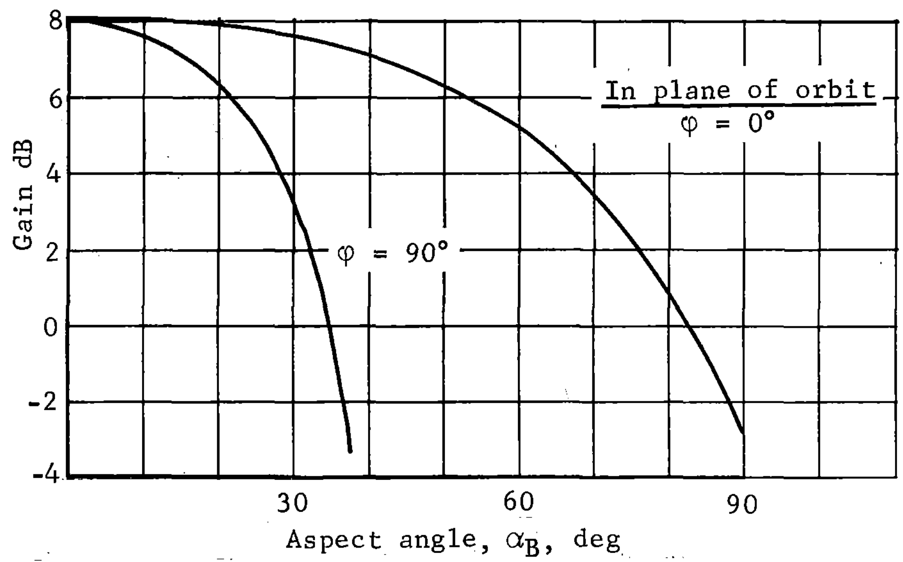


Figure 41.- Spacecraft Relay Antenna Power Gain Pattern

Variations in the spacecraft antenna aspect angle establish the required spacecraft antenna beamwidth in the plane of the orbit. Figures 38 and 41 show the aspect angle variations and the assumed spacecraft antenna beamwidth, respectively. The reference spacecraft antenna boresight was selected to provide near-optimum link performance at BVS/aeroshell separation, yet provide adequate coverage to satisfy the initial contact view-time requirements. Further analysis of subsequent contacts, to be encountered in postdeploy operations, has shown an alternative spacecraft antenna boresight is required. This alternative boresight requires a single axis offset of approximately 50° in the orbit plane. Since shifting the antenna boresight for subsequent passes is required, and since this shift also improves the margin near the end of the initial pass, the shift can take place during the initial pass.

System performance was evaluated for the two extreme atmosphere models. The most significant effect of the extreme atmosphere models on the communications link between the spacecraft and BVS/entry vehicle occurs near entry and at subsonic probe impact. The time history of the BVS/entry vehicle attitude accounts for the difference in performance margin near entry. If the dense atmosphere model is encountered, the BVS/entry vehicle will approach the radial descent attitude earlier resulting in increased antenna pointing loss and lower margins. Once the BVS/entry vehicle assumes the radial descent attitude for both atmosphere models ($\approx$100 sec after entry), the performance margin will be independent of the atmosphere encountered. The time from entry to subsonic probe impact will be approximately 100 sec longer for the dense atmosphere model.

The system performance margin will vary with time and with the atmosphere encountered, as shown in figures 42 and 43. The performance margin shown was derived for the nominal entry conditions noted. The sample design control table (table 26) demonstrates the method of performance margin calculation. The BVS and subsonic probe communication systems are assumed identical from the standpoint of effective radiated power, information bit rate, and modulation. Therefore, a single profile can be used to represent the performance of both systems. The fact that the BVS will float at approximately 58 km altitude while the subsonic probe descends to the surface will not affect the geometry significantly since the communication range to the spacecraft is much greater than the difference between the BVS and subsonic probe altitudes. Reorientation of the spacecraft antenna to a boresight compatible with future postdeploy contacts, before reaching the terminal elevation mask of the initial contact, will increase the view time by approximately 7 minutes, as shown in figure 43.

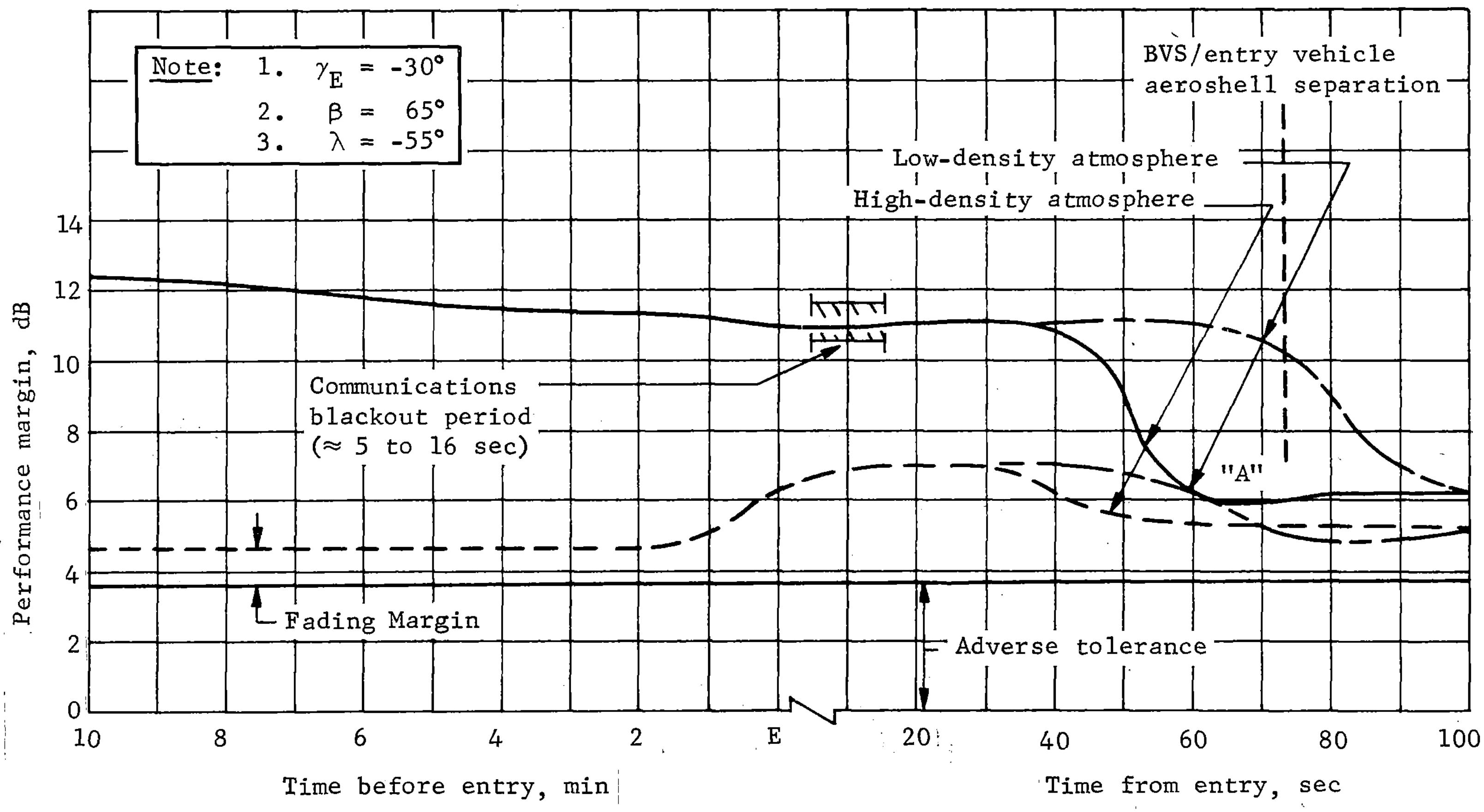


Figure 42.- Performance Margin Profile (Entry - 10 min thru Entry + 200 sec)

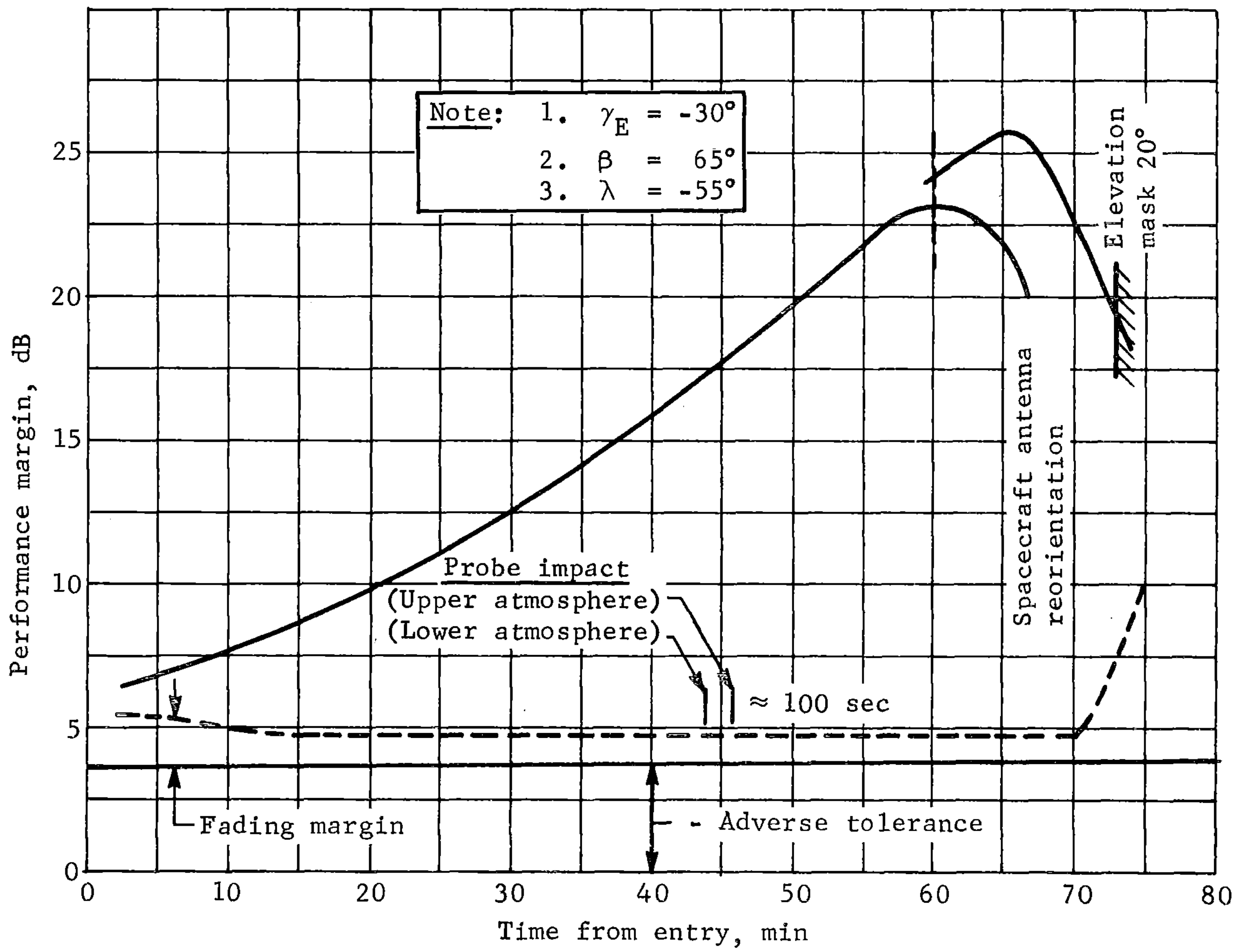


Figure 43.- Performance Margin Profile (Entry + 200 sec thru Initial Contact Terminal Elevation Mask)

TABLE 26.- 1973 ENTRY FROM ORBIT RELAY LINK PERFORMANCE, TELECOMMUNICATIONS DESIGN CONTROL TABLE

No.	Parameter	Value	Tolerance (+)	Tolerance (-)	Notes
1	Transmitter power	+43.0 dBm	0.2	0.0	20 W min.
2	Transmitting circuit loss	-1.0 dB	0.2	0.3	
3	Transmitting antenna gain	+4.8 dB	0.5	0.3	
4	Transmitting antenna pointing loss	-5.2 dB	----	----	Max α_p = 70°
5	Space loss F = 390 MHz, ρ_c = 21 011 km	-170.7 dB	----	----	
6	Polarization loss	-0.3 dB	0.2	0.2	
7	Ionosphere attenuation	-0.7 dB	0.6	0.7	
8	Receiving antenna gain	+8.0 dB	0.5	0.5	
9	Receiving antenna pointing loss	-0.6 dB	----	----	Boresight α_{B1} = -110° α_B = -23°
10	Receiving circuit loss	-0.5 dB	0.0	0.7	
11	Net circuit loss	-166.2 dB	2.0	2.7	
12	Received carrier power	-123.2 dBm	2.2	2.7	
13	Receiver noise spectral density T system = 786°K	-169.6 dBm/Hz	1.0	1.0	1000°K max
14	Predetection noise bandwidth bit rate 240 bps;	+44.3 dB-Hz	----	----	27 kHz
15	Receiver noise power	-125.3 dBm	1.0	1.0	
16	Carrier-to-noise ratio	+2.1 dB	3.2	3.7	
17	Threshold carrier-to-noise ratio $P_e^b = 4 \times 10^{-3}$, TW = 112	-3.9 dB	----	----	
18	Fading allowance	----	----	2.0	
19	Performance margin	+6.0 dB	3.2	5.7	

Note: Calculations for Point "A" (fig. 42)

The fading margin required to overcome the slow fading environment is also shown on the performance margin profiles of figures 42 and 43. The analysis performed to determine the required fading margin considered the planet surface characteristics, the incident angle of the reflected signal on the planet surface, the BVS antenna gain distribution, and the differential path length between the direct and reflected signals. The time delay between the direct and reflected signals is not a factor in defining the fading environment since it represents only a small fraction of the reference system information bit period. The method of determining the required fading margin is discussed in appendix G.

Effect of targeting parameter on entry communications: It is of interest to determine the effect on the entry relay telemetry link of varying the targeting parameter, and conversely to determine the constraint on targeting due to the radio subsystem.

The angle, β, measured clockwise from the orbit periapsis to the point at which the BVS/entry vehicle enters the Venus atmosphere is defined as the targeting parameter. The variation in targeting parameter, $\Delta\beta$, that can be achieved without violation of established system constraints is a measure of the targeting capability. The system constraints and parameters that define the entry conditions are shown in figure 44.

The effect of entry parameter variations on the communication system performance is shown in figure 45. The targeting parameter, β, is constrained to upper and lower limits by the angle of attack and the deorbit velocity, respectively. The associated spacecraft lead angle, λ, is constrained on the left by the required view time and on the right by the elevation mask. Constant view-time constraint lines are shown for view times of 30 to 45 minutes. Compliance with these system constraints requires that the entry parameters be selected from within the constraint lines.

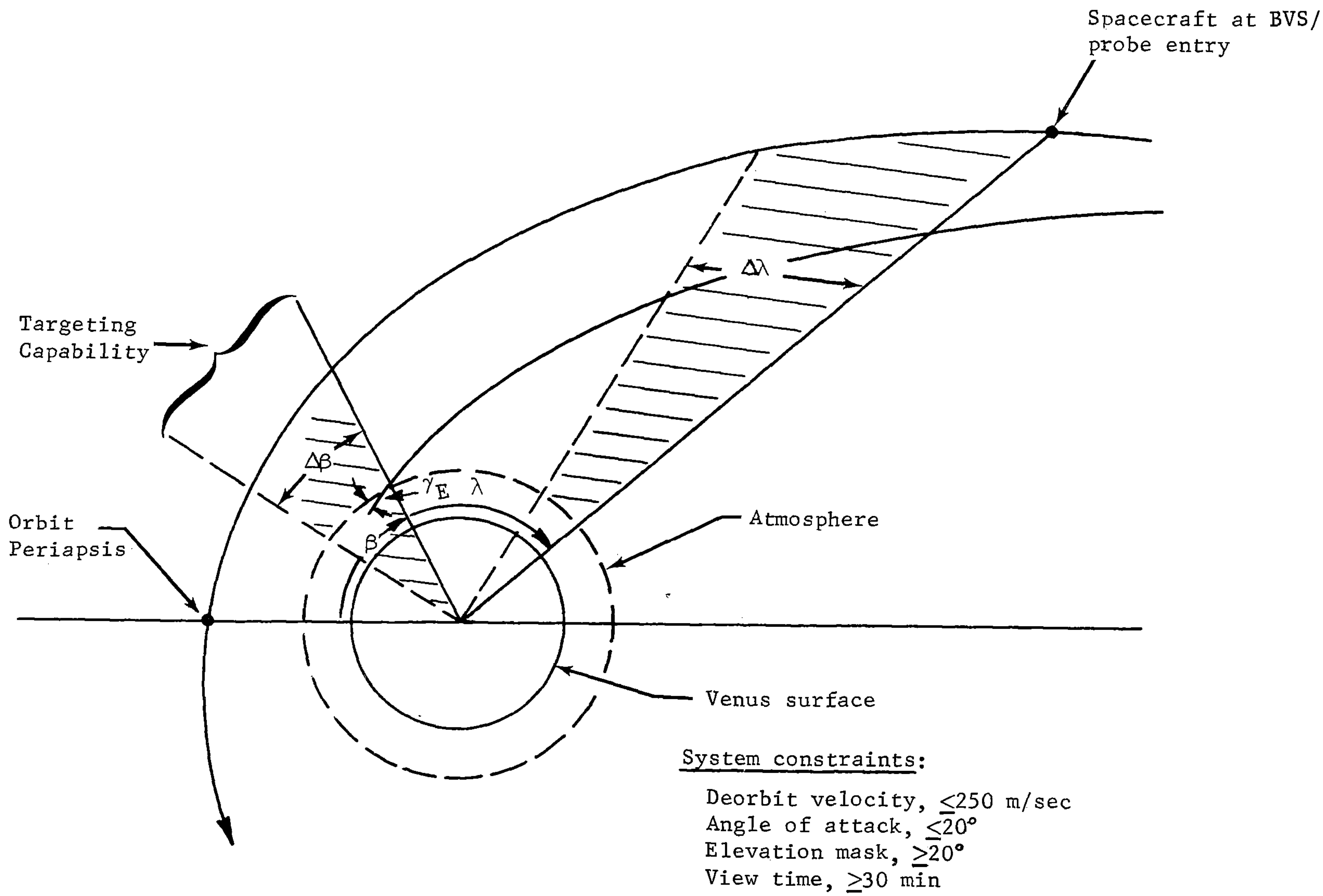


Figure 44.- Entry Geometry

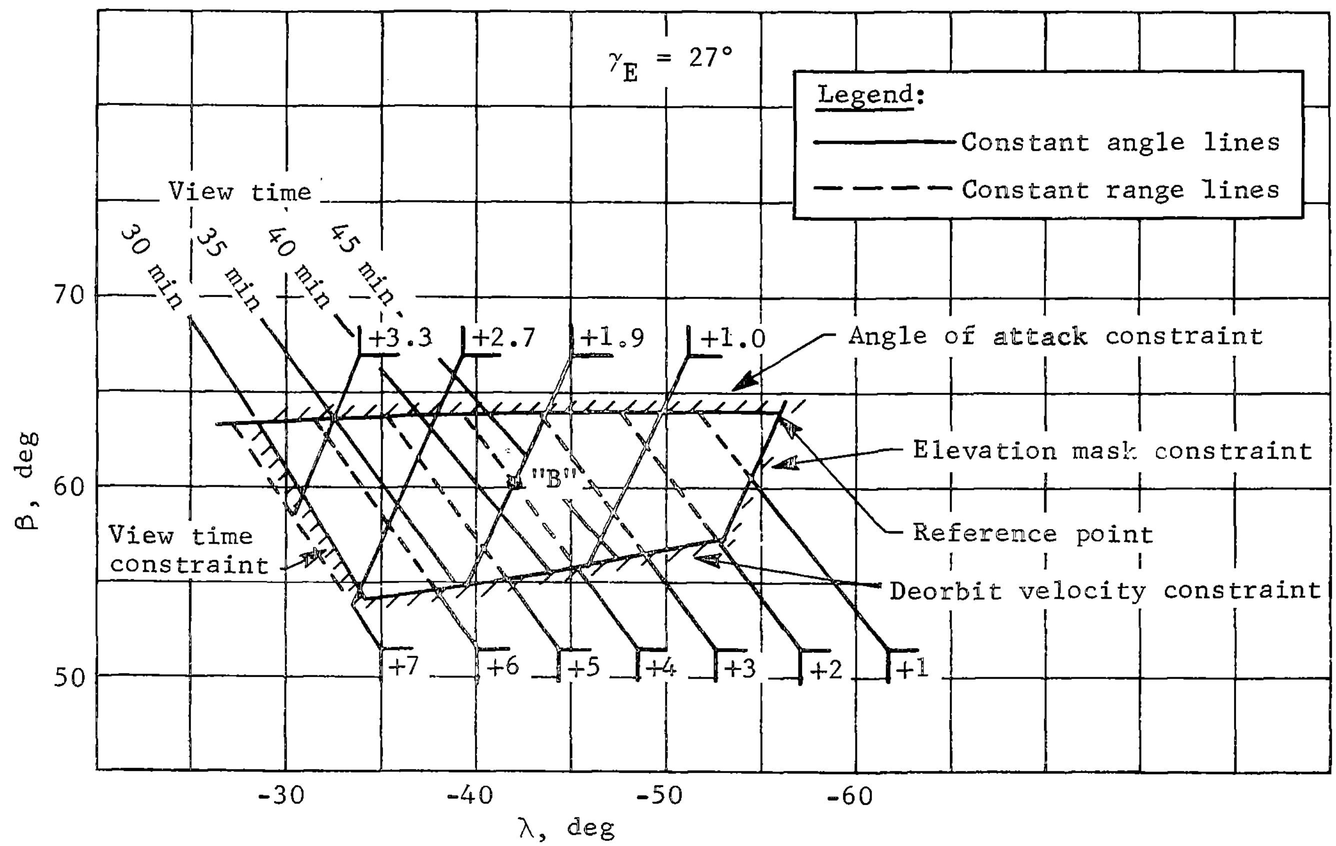


Figure 45.- Entry Parameters (γ_E = -27°)

The point in the deorbit trajectory at which the minimum performance margin occurs (point "A", fig. 42) is represented by the reference point noted in figure 45. The specific communications geometry associated with point "A" are a BVS antenna aspect angle of 70° and a communication range of 21 000 km. The spacecraft antenna will be boresighted to the BVS/entry vehicle near this point in the entry phase, to optimize the minimum system performance margin. Therefore the spacecraft antenna aspect angle will be independent of the entry conditions. The constant range lines noted on figure 45 represent entry parameters, β and λ, for which constant communication range occurs. A number is associated with each constant range line. The number represents the improvement in decibels associated with a decrease in space loss. The decreased space loss results from limiting the communication range to some value less that 21 000 km at the minimum system performance margin point. Similar constant angle lines were derived and shown as solid lines within the system constraints. The number associated with each constant angle line represents the decibel improvement in BVS antenna pointing loss. The decrease in pointing loss can be obtained by limiting the BVS antenna aspect angle to some value less than 70° at the minimum system performance margin point. With this in mind, the impact of variations in entry parameters on the minimum system performance margin can be determined. For example, the reference point represents a minimum system performance margin of 0.3 dB for a targeting parameter, β, of 64° and a spacecraft angle, λ, of -56°. Suppose a β of 60° and a λ of -42° were the desired entry parameters (point "B"). A 1.9 dB improvement in BVS antenna gain is associated with the constant angle line and a 4 dB improvement in space loss is associated with the constant range line. Therefore, the total improvement is 5.9 dB and the minimum system performance margin for those entry parameters will be 6.2 dB. Figure 45 was derived from data obtained for an entry flight path angle, γ_E, of -27°. Data derived for entry flight path angles of -27°, -30°, and -33° are shown in figure 46. It can be seen that with proper selection of entry parameters, $(\gamma_E, \beta, \lambda)$, a targeting capability, $\Delta\beta$, of 15° can be obtained without degrading the minimum system performance margin.

BVS to spacecraft radio link performance (postdeploy operations): Since the BVS is free to drift with the wind as it floats in the atmosphere of Venus, it is important to determine the communications time available for transmission of telemetry data to the relay spacecraft for various locations of the BVS in relation to the subperiapsis point and the orbital plane of the orbiter.

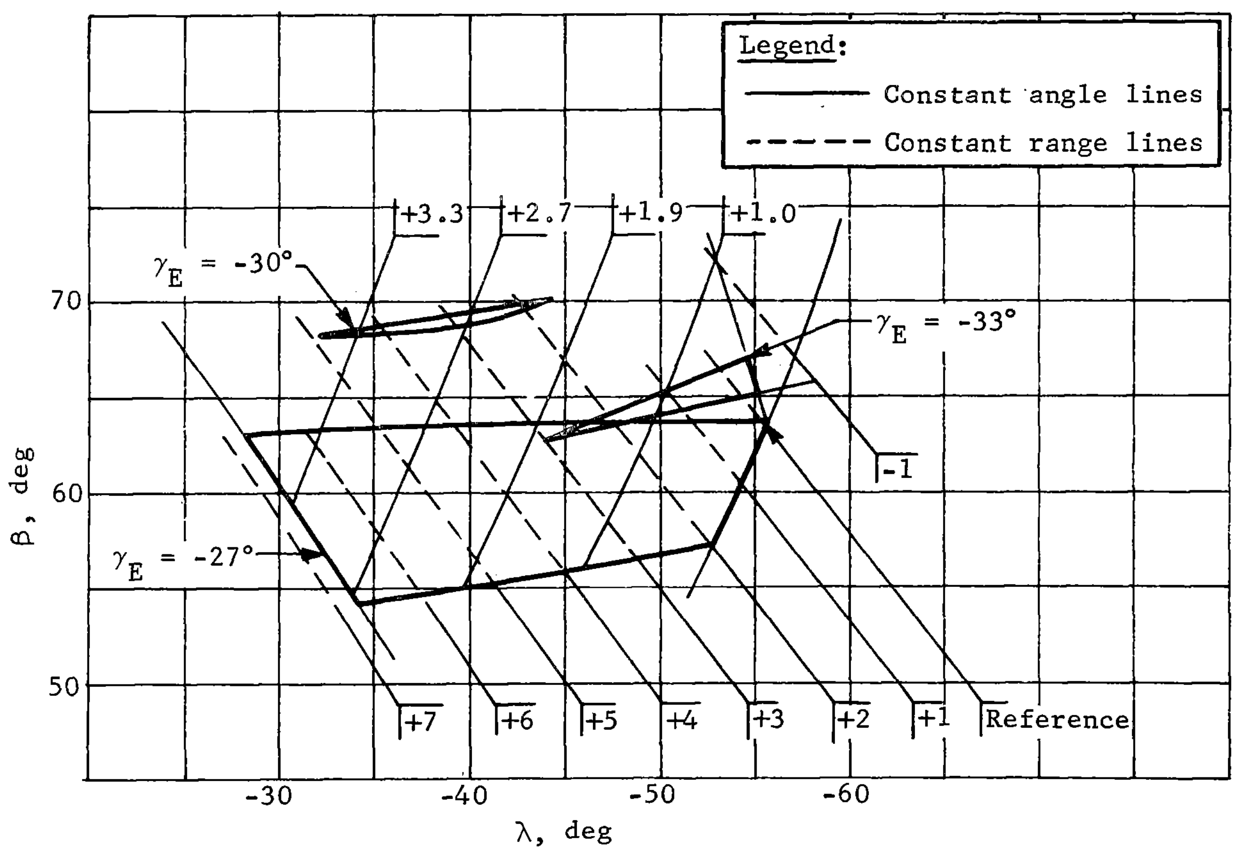


Figure 46.- Entry Parameters (-27° > γ_E > -33°)

Station positions in the plane of the orbiter's orbit from the subperiapsis point (0° longitude) to 120° from the subperiapsis point in increments of 10° and for the same longitudes, but for out-of-plane locations (latitudes) of 10°, 20°, 30°, 40°, and 50° were analyzed.

The analysis was done by computer to calculate the gometry for each pass and the communications link margin variation versus time for each station location as the orbiter moved in and out of optical line of sight. Included in the link margin versus time computations are range, antenna gains as a function of aspect angles, multipath allowance, as well as fixed gains and losses.

Antenna patterns and sample design control table calculations are as given in the previous section. The method of determining the effect of multipath is discussed in appendix G.

Boresighting of the spacecraft antenna (which is fixed in relation to spacecraft axes once the station deployment is complete) for purposes of this analysis was done graphically without the benefit of iteration of the results. Therefore the boresight may be less than optimum, but it is representative of results that can be obtained.

The boresight chosen was one in which the spacecraft antenna axis points to the center of the planet when the spacecraft is at true anomaly of 300°. The wider beamwidth of the elliptical pattern is in the plane of the orbit. This gives good results when attempting to cover a station drift area centered at the baseline mission entry point at a latitude of 65° and longitude of 0° as related to the orbiter's orbit.

It is impractical to plot the variation versus time of all the link parameters for each of the more than 60 station locations considered. The significant results of the investigation are shown in figure 47, which gives a family of curves showing length of time (for various station locations) in which the baseline telemetry link margins are above 0 dB for a bit error rate of 4×10^{-3}.

Any location showing 5 minutes or more of positive margin provides adequate time for one full transfer of data to the spacecraft. A period of 10 minutes gives time for one full repeat of all data stored in the BVS memories, as well as several additional frames of real-time data.

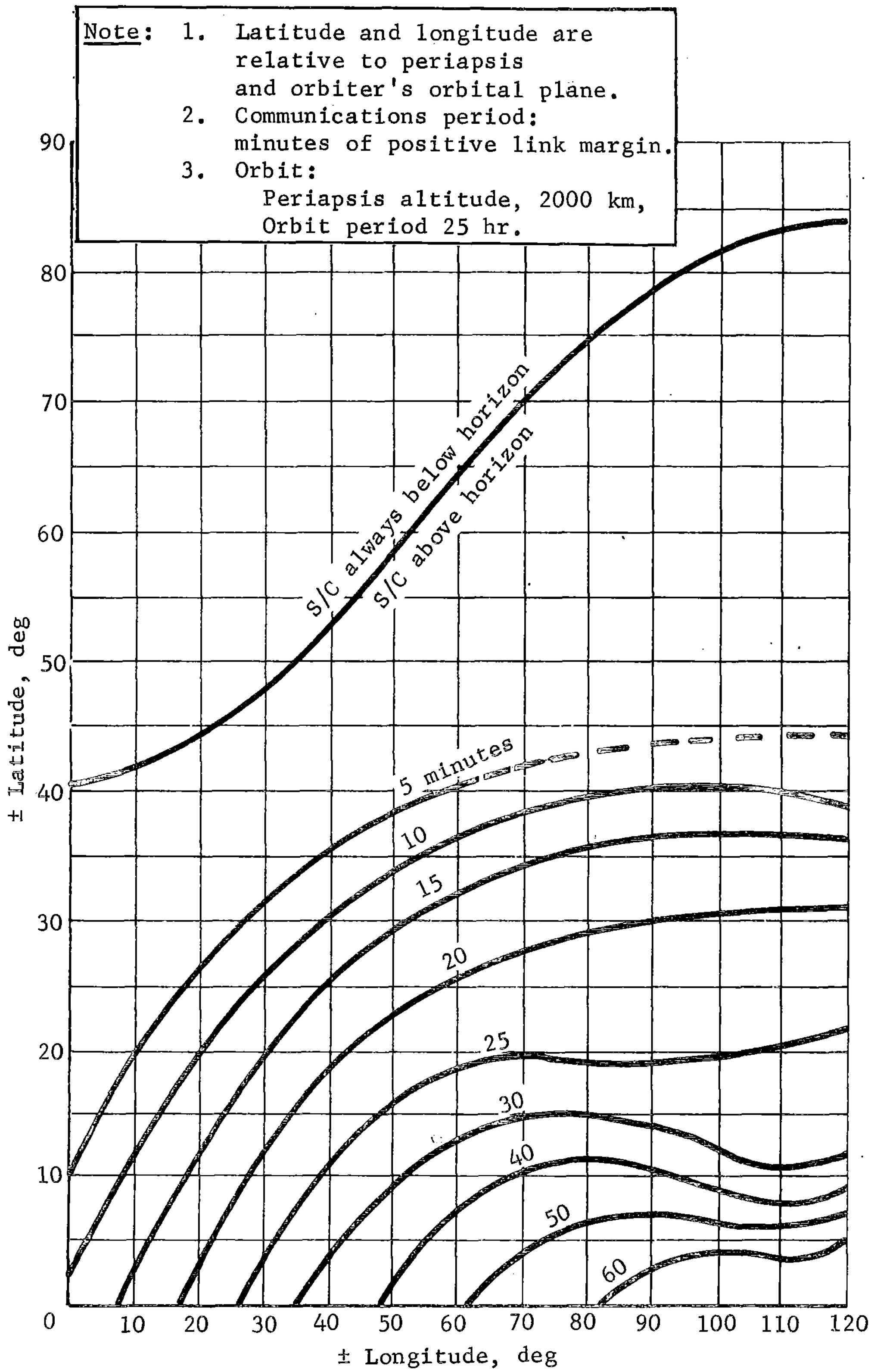


Figure 47.- Communication Period vs Station Position

At least 5 minutes of communications time is available for locations within about 40° of the initial entry site assuming an entry at a true anomaly of 65°. Allowable drift of the station in the plane of the orbit and away from periapsis is greater than 60°.

The system margin versus time for one typical station location is shown in figure 48 along with the adverse tolerance and variation in multipath. Figure 49 shows antenna gain variations, change in communication range, and spacecraft elevation angle versus time for the same pass. The beginning point for the pass was arbitrarily chosen since "turn-on" of the BVS transmitter is controlled by the command link threshold. Note that telemetry transmission occurs during only the first 10 minutes after which the BVS transmitter is turned off. The system has no way of knowing that delay of the beginning of the transmission would have provided better margins later for a full 10-minute transmission because the available view time is unknown due to lack of knowledge of where the station has drifted in relation to the orbit of the spacecraft.

Mechanization options: Several options were considered in selecting the baseline approach for the command and telemetry communications links to the spacecraft.

During the entry phase, entry data could be transmitted back to the spacecraft via the BVS transmitter, the subsonic probe transmitter, or a separate transmitter in the aeroshell.

Because of the need to "bury" the subsonic probe in the aeroshell to obtain a good BVS/entry vehicle configuration, and to prevent redesign of the BVS communications to handle entry data in case the subsonic probe concept was deleted, it was decided that the BVS transmitter and data subsystem would be used in the transmission of entry data to the relay spacecraft.

Other major decisions were made in selecting the frequency and modulation for the telemetry and command link and in choosing an antenna for the spacecraft.

Selection of frequencies in the 400 MHz range was a compromise. For lower frequencies (200 or 300 MHz) excessive ionospheric losses and adverse tolerances were predicted. Use of higher frequencies results in excessive space loss (small antenna aperture) since required antenna beamwidths are dictated by the geometry of the problem and remain fixed regardless of the frequency.

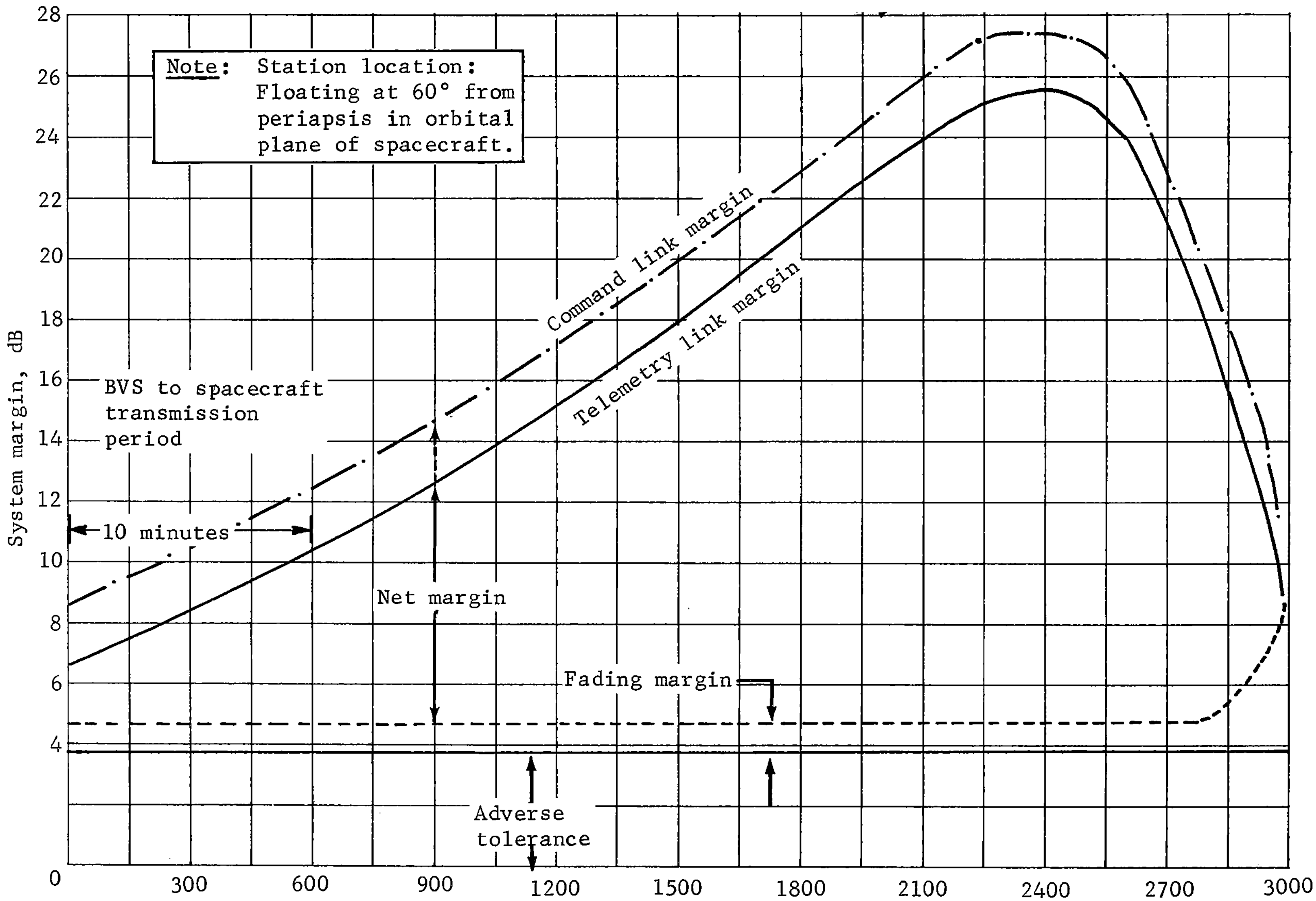


Figure 48.- System Margin vs Time

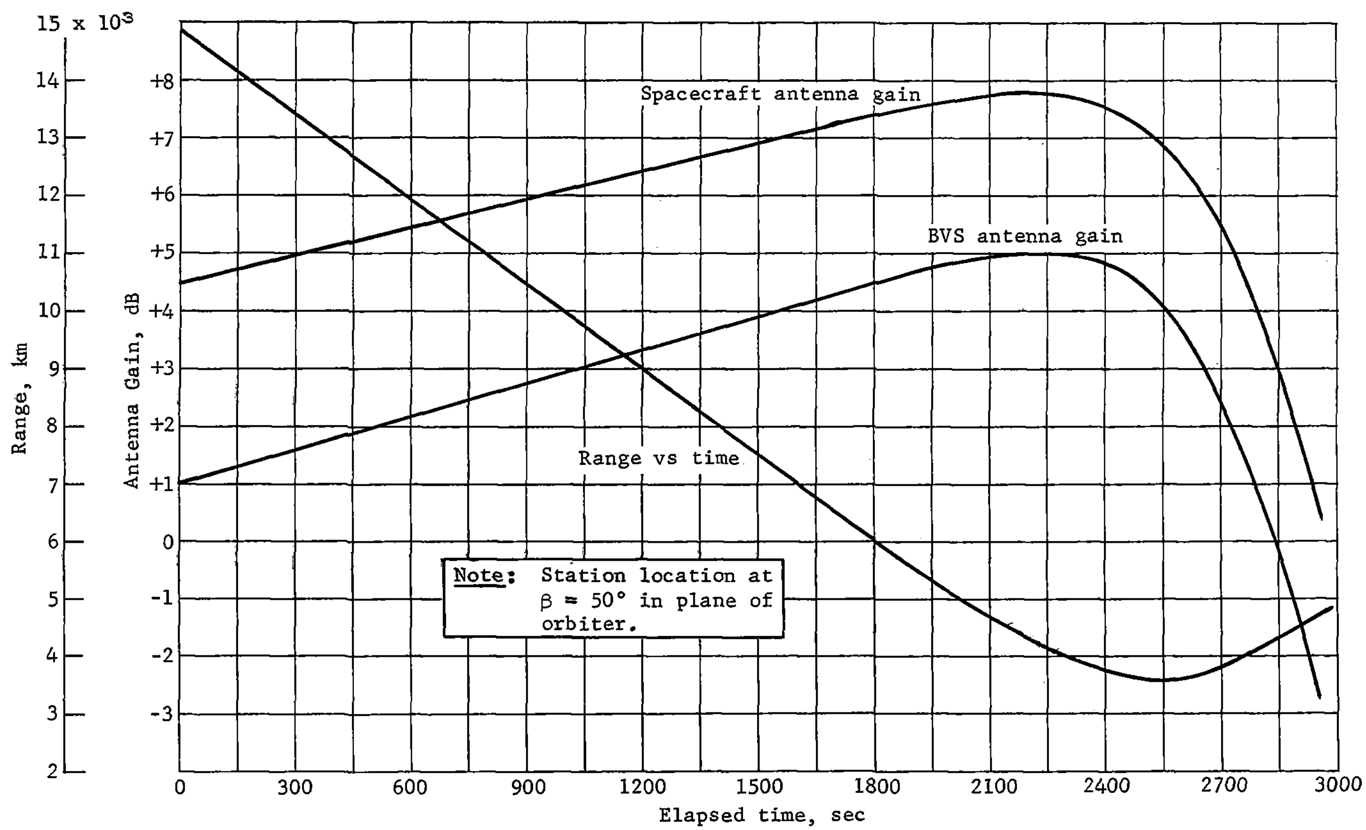


Figure 49.- Range and Antenna Gain vs Time

For example, ionospheric attenuation at 200 MHz is predicted to be 2.95 dB with an uncertainty of plus 2.95 dB minus 2.65 dB (refs. 5 and 6), while for 200 MHz the loss is estimated at 0.74 dB with a uncertainty of plus 0.74 dB to minus 0.66 dB. The ionospheric loss and adverse tolerance at 200 MHz completely counterbalance the gains one would get from the 6 dB lower space loss. Therefore, one can use the 400 MHz frequency without losing power and yet gain the physical advantage of a smaller antenna (smaller than at 200 MHz).

Limitations in the maximum diameter of the subsonic probe restrict the lower end of the frequency range to about 400 MHz. By using the 400 MHz range for both BVS and subsonic probe, common equipment can be used.

Selection of noncoherent frequency shift key (FSK) modulation over coherent PSK was based strictly on the desire to avoid use of unattended automatic frequency search modes in both the spacecraft and in the BVS. The search modes would be required due to the large frequency uncertainties associated with doppler and oscillator instabilities of the transmitters and receivers.

A benefit of commonality of equipment was realized by use of noncoherent FSK for all relay links. The three relay transmitters (one each in the BVS, spacecraft, and subsonic probe) can all be identical units. The spacecraft and BVS relay receivers can also be made identical. The crystal frequencies of course must be different for each.

A penalty of approximately 5 dB in link margin is paid for going to FSK, but it is felt that the benefits in commonality of subsystems as well as elimination of the frequency search modes are more than offsetting for the low data rate requirements of this mission.

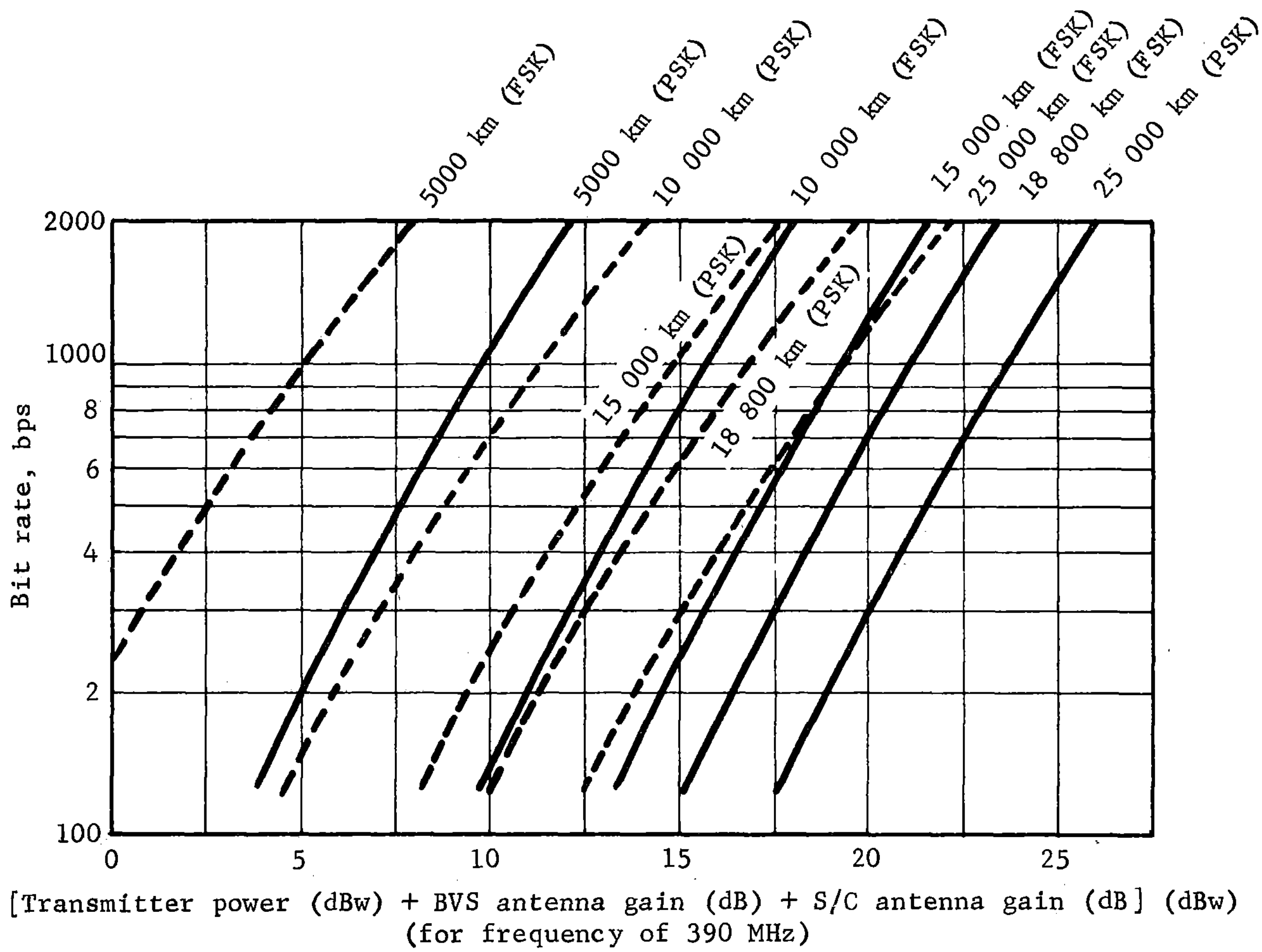
Figure 50 compares the bit rate capability versus transmitter power antenna gain products for various communications ranges for both coherent PSK/PM and noncoherent FSK. A bit error rate of 4×10^{-3} is assumed. Examples of calculations used in making this comparison are shown in tables 27 and 28.

Legend:

—— Noncoherent PCM/FSK

- - - - Coherent PCM/PSK/PM

Note: 1. See reference calculations for bandwidths and other parameters.
2. For bit error rate 4×10^{-3}.



Example: For 20 W transmitter, 8 dB S/C antenna gain + 0 dB BVS antenna gain minus 3 dB for multipath allowance gives 18 dBw. For FSK modulation, data rates of 370 or less can be used for ranges to 18 800 km assuming no antenna pointing loss. For PSK/PM modulation, data rates up to 1400 bps may be used. Actual combinations of range, realized antenna gains, and multipath are a function of the orbit and station position. However, for any given combination, a comparison between FSK and PSK systems can be made.

Figure 50.- Data Rate vs Transmitter Power Antenna Gain Products for Various Communication Ranges and Types of Modulation

TABLE 27.- BVS-TO-ORBITER REFERENCE LINK CALCULATION (FSK, 128 bps, 18 800 km, 390 MHz)

Parameter	Value	Tolerance +	Tolerance -
Total transmitter power	+43 dBm	0.2	0.0
Polarization loss	-0.3 dB	0.2	0.2
Ionospheric attenuation	-0.7 dB	0.6	0.7
Transmitter circuit loss	-1.0 dB	0.2	0.3
Transmitter antenna gain	+4.8 dB	0.5	0.3
Transmitter antenna point loss	-4.8 dB		
Space loss (range 18 800 km, frequency 390 MHz)	-169.8 dB		
Receiving antenna gain (on axis)	+8.0 dB	0.5	0.5
Receiving antenna point loss	-8.0 dB	----	----
Receiving circuit loss	-0.5 dB	0.0	0.7
Net losses	-172.3 dB	2.2	2.7
Total received power	-129.3 dB	2.2	2.7
System noise spectral density	-169.6 dBm	1.0	1.0
Channel filter bandwidth (27 000)	+44.3 dB		
Net noise power	-125.3 dBm	1.0	1.0
Carrier-to-noise ratio	-4.0 dB	3.2	3.7
Required S/N (in predetection BW) ($P_e = 4 \times 10^{-3}$, TW = 210, 128 bps)	-5.5 dB		
Performance margin	+1.5 dB (see notes)	3.2	3.7

Note: 1. No antenna gains are shown for reference calculation (pointing loss = antenna gain).

2. Difference between performance margin and sum of adverse tolerance plus multipath allowance must be made up by antenna gains and/or range reduction.

3. Baseline system operates at a data rate of 240 bps.

TABLE 28.- REFERENCE BVS TELEMETRY PSK/PM (128 bps, 390 MHz, 20 W TRANSMITTER, 18 800 km RANGE)

Parameter	Value	Tolerance	
		+	-
Total transmitter power	+43 dBm	0.2	0.0
Polarization loss	-0.3 dB	0.2	0.2
Ionospheric attenuation	-0.7 dB	0.6	0.7
Transmitter circuit loss	-1.0 dB	0.2	0.3
Transmitter antenna gain (on axis)	+4.8 dB	0.5	0.3
Transmitter antenna pointing loss	-4.8 dB		
Space loss (range 18 800 km, frequency 390 MHz)	-169.8 dB		
Receiving antenna gain (on axis)	+8.0 dB	0.5	0.5
Receiving antenna pointing loss	-8.0 dB		
Receiving circuit loss	-0.5 dB	0.0	0.7
Net losses	-172.3 dB	2.2	2.7
Total received power	-129.3 dBm	2.2	2.7
System noise spectral density	-169.6 dBm	1.0	1.0
Carrier channel			
Carrier modulation loss	-3.5 dB		
Received carrier power	-132.8 dBm	2.2	2.7
Carrier APC noise bandwith ($2\ B_{LO} = 160$ Hz)	-22.05 dB		
Threshold S/N in $2\ B_{LO}$	+8.0 dB		
Threshold carrier power	-139.55 dBm	1.0	1.0
Performance margin	+6.7 dB	3.2	3.7
Data channel			
Modulation loss	-2.57 dBm		
Received data subcarrier power	-131.9 dBm	2.2	2.7
Bit rate (128 bps)	+21.07 dB		
Required St/N/B	+9.0 dB		
Circuit loss	-1.06 dB		
Threshold subcarrier power	-138.47 dBm	1.0	1.0
Performance margin	+6.57 dB (see note)	3.2	3.7

Note:
1. No antenna gains are shown (pointing loss = antenna gain).
2. Difference between margin and sum of adverse tolerance + multipath must be made up by antenna gains and/or range reduction.
3. This is an example of calculations used in making comparisons between PSK and FSK modulation. The baseline system used FSK modulation at a data rate of 240 bps.

Coding for the relay link was not seriously considered because of the desire to minimize spacecraft additions and modifications. Should the data rate requirements be drastically increased, as would occur if imaging sensors were added (TV for example), a coherent link with coding would be a prime contender. The impact of imaging on the BVS telecommunications subsystems is discussed in appendix D.

An investigation of postdeploy communications coverage for three types of spacecraft antennas determined whether the elliptical pattern was indeed better than either an omni or a symmetrical pattern. The computer runs confirmed that the elliptical pattern provided the better coverage for station drifts up to 40° in all directions from the entry location and that beneficial gain from the antenna was realized.

Options to provide two-way doppler measurements are discussed in appendix H.

Communications for the entry-from-approach mode were investigated for the orbital mission as well as for entry from orbit. The entry geometry and performance margins for entry from approach are nearly identical with those of the flyby mission and therefore are not presented. See the Venus flyby discussion.

Command subsystem. - Functions of the command subsystem are to detect and decode command signals received from the spacecraft and to initiate discretes to the appropriate subsystems in response to the transmitted command.

A list of commands is given in table 29. All are real-time commands requiring momentary output pulses to initiate action.

TABLE 29.- COMMAND LIST

1.	Release drop sonde (after fixed delay).
2.	Transfer to low-power mode.
3.	Transfer to high-power mode.
4.	Turn on transmitter.
5.	Turn on transmitter (alternative command channel).
6.	Begin doppler measurement.

Requirements and description: Command requirements are shown in table 30. The command subsystem is typically made up of a simple sequential tone decoder requiring an "alert" tone to activate the unit, followed by two command tones in sequence to activate an output switch. A single alert tone and three command tones can be used to provide a six command capability. Receipt of the alert tone activates the decoding circuitry only long enough to receive the command tones. This reduces the probability of a decoded output due to noise. Use of an "exclusive or" gate, and integrating networks can also be used to guard against decoding due to noise. A weight of 1.5 lb and 3.5 W of power have been allotted for the decoder.

TABLE 30.- COMMAND SUBSYSTEM REQUIREMENTS

1.	Capacity to decode six commands.
2.	Activate decoding circuitry on receipt of an alert tone.
3.	Decode three sequential tone combinations following the "alert" tone.
4.	Have immunity to impulse and broadband noise.

Performance: A reference design control table for the command link is shown in table 31. The calculations are based on use of the BVS receiver described in the radio subsystem section and tone filter bandwidths of 10 Hz in the decoder. A 20 W spacecraft transmitter is frequency shift keyed at the rates required to generate the command tones. Link margin versus time for a typical postdeploy communication pass is shown in figure 48 along with that of the telemetry link. The system margin is 1.8 dB above that of the telemetry link.

Mechanization options: Two types of command systems have been considered for the spacecraft-to-BVS link -- a tone system described above and a binary PCM system. Frequency shift key modulation has been chosen regardless of tone or PCM so that the spacecraft transmitter and BVS receiver are of the same basic design as the BVS-to-spacecraft telemetry link. Only the decoder is different for tone as compared to PCM.

For the few commands required, a tone decoder is lighter in weight than a binary PCM decoder and less complicated.

TABLE 31.- REFERENCE LINK CALCULATION COMMAND LINK (FSK TONE MODULATION FREQUENCY 410 MHz)

Parameter	Value	Tolerance (db)	
		+	-
Total transmitter power	+43.0 dBm	0.2	0.0
Polarization loss	-0.3 dB	0.2	0.2
Ionosphere attenuation	-0.7 dB	0.6	0.7
Transmitter circuit loss	-1.0 dB	0.2	0.3
Transmitter antenna gain	+8.0 dB	0.5	0.5
Transmitter antenna pointing loss	-8.0 dB	----	----
Space loss (range 18 800 km, 410 MHz)	-170.2 dB		
Receiving antenna gain (on axis)	+4.8 dB	0.5	0.3
Receiving antenna pointing loss	-4.8 dB	----	----
Receiving circuit loss	-0.5 dB	0.0	0.7
Net losses	-172.7 dB	2.2	2.7
Total received power	-129.7 dBm	2.2	2.7
System noise spectral density	-169.6 dBm	1.0	1.0
Channel filter bandwidth (27 000)	+44.3 dB		
Net noise power	-125.3 dBm	1.0	1.0
Carrier-to-noise ratio	-4.4 dB	3.2	3.7
Required $S/N_{in} = a + \sqrt{a(a+2)} = -6.2$ dB where $a = S/N_{out}\ (1/TW) = .0233$ and $TW = \dfrac{\text{channel filter BW}}{\text{tone filter BW}} = 2700$	-6.2 dB	where: S/N_o =	req'd +18 dB
Performance margin	+1.8 dB (see note)	3.2	3.7
Doppler tracking loop			
System noise spectral density	-169.6 dBm	1.0	1.0
APC bandwidth (160 Hz)	+22.05 dB		
Threshold SNR in 2 B_{LO}	+12.00 dB		
Threshold carrier power	-135.55 dBm		
Received carrier power	-129.5 dBm	2.2	2.7
Performance margin	+6.05 dB (see note)	3.2	3.7

Note: 1. No antenna gain is shown for the reference calculations (pointing loss = antenna gains).
2. Difference between performance margin and sum of adverse tolerance plus multipath is made up by antenna gains and/or range reduction for actual geometry.

Data subsystem. - Table 32 lists the engineering and science data requirements for the BVS. The measurements are grouped into formats depending on the time period they are required. The sampling rate of each measurement is shown for the format period collected. Table 33 lists the data formats and the periods of time that they are used. The required sampling rates result in a data rate of 240 bps.

The data subsystem provides all the sequencing control associated with the collection, processing and transmission of data on the BVS. This includes sequencing of science experiments, data format control, and transmitter control. Table 34 lists the data subsystem sequencer requirements for the period of capsule spacecraft separation. Entry and deployment sequence requirements are shown in Table 35.

Subsystem description: The data subsystem consists of a data multiplexer/encoder, memory, transducer power supply, and a data subsystem sequencer, all located in the BVS. A remote multiplexer and a transducer power supply are located in the aeroshell. A block diagram of the subsystem is shown in figure 51. All science and engineering data are processed through this data subsystem.

The data subsystem sequencer (DSS) provides all the timing and power control required on the BVS for the sequencing of experiments, data subsystem equipment, command, and radio equipment to collect, process, store, and transmit data. It operates continuously from BVS entry until end of mission. Table 36 is a list of discretes issued by the DSS.

The data multiplexer/encoder accepts 0-40 MV and 0-5 Vdc analog signals, bilevel and digital signals, provides a digital signal interface with the memory, and provides control of the aeroshell multiplexer. A hybrid level signal input for analog signals is selected to minimize the signal conditioning and the transducer power requirements. The format is generated by this unit for those data that are sampled at regular intervals from various instruments. Experiments with unique data handling requirements such as the gas chromatograph, mass spectrometer, sonde, accelerometer for monitoring wind gusts, and the position determination have their own associated data processing electronics and provide a digital data interface with the data multiplexer/encoder. Formatting and buffer storage of an experiment sample is accomplished in each experiment data unit. Clocking and control is provided by the data subsystem sequencer. Data from the various experiment buffers is sequenced into the data memory under control of the DSS.

TABLE 32.- ENGINEERING AND SCIENCE DATA REQUIREMENTS

Measurement	Signal level	Format (sps)					
		I	A	B	C	D	E
Temp, aeroshell 1	0-40 MV	x	½	½	½		
Temp, aeroshell 2	↓	↓	½	↓	↓		
Temp, aeroshell 3	↓	↓	½	↓	↓		
Temp, backshell 1	↓	↓		↓	↓		
Temp, backshell 2	0-40 MV	x		↓	↓		
Ablator thickness 1	3 bit digital			↓	↓		
Ablator thickness 2	↓			↓	↓		
Ablator thickness 3	↓			↓	↓		
Ablator thickness 4	↓			↓	↓		
Ablator thickness 5	3 bit digital			↓	↓		
Temp, gondola 1	0-40 MV	x		↓	↓	½	1/hr
Temp, gondola 2	↓	↓		½	½	½	1/hr
Temp, fuel tank	0-40 MV	↓	½				
Press., fuel tank	0-5 V	↓	↓				
Temp, oxidizer tank	0-40 MV	↓	↓				
Press., oxidizer tank	0-5 V	↓	↓				
Temp, gas tank	0-40 MV	↓	↓				
Press., gas tank	0-5 V	↓	½				
Temp, inflation tank	0-40 MV	↓		½	½	½	
Press., inflation tank	0-5 V	↓		↓	↓	↓	
Temp, Ag-Zn battery, BVS	0-40 MV	↓		↓	↓	↓	1/hr
Volt, Ag-Zn battery, BVS	0-5 V	↓		½	½	½	1/hr
Current, charging Ag-Zn battery, BVS	0-40 MV	↓					1/hr
Temp, probe battery	0-40 MV	↓		½	½		
Volt, probe battery	0-5 V	↓		½	½		
Current, charging probe battery	0-40 MV	↓					
Temp, aeroshell battery	0-40 MV	↓	½	½	½		
Volt, aeroshell battery	0-5 V	↓	½	½	½		
Current charging aeroshell battery	0-40 MV	↓					
Volt, transducer power supply (aeroshell)	0-5 V	x	½	½	½		
Volt, transducer power supply (BVS)	0-5 V		↓	↓	↓	½	1/hr
Volt, BVS power bus	0-5 V		↓	↓	↓	↓	↓
Current, BVS power bus	0-40 MV		↓	↓	↓	↓	↓
Power, BVS transmitter output	0-40 MV		↓	↓	↓	↓	↓
Temp, transmitter	0-40 MV		½	½	½	½	1/hr

TABLE 32.- ENGINEERING AND SCIENCE DATA REQUIREMENTS - Continued

Measurement	Signal level	Format (sps)					
		I	A	B	C	D	E
Press., deflection motor thrust chamber	0-5 V		10				
Accelerometer, long, low range	0-40 MV		5	1	1		
Time, 8 bit	8 bit digital		½	½	½	½	
Time, 16 bit	16 bit digital						1/hr
Discrete, aeroshell sequencer 1	Bilevel		½	½	½		
Discrete, aeroshell sequencer 2	↓		↓	↓	↓		
Discrete, aeroshell sequencer 3	↓		↓	↓	↓		
Discrete, aeroshell sequencer 4	↓		↓	↓	↓		
Discrete, aeroshell sequencer 5	↓		↓	↓	↓		
Discrete, aeroshell sequencer 6	↓		↓	↓	↓		
Discrete, aeroshell sequencer 7	↓		↓	↓	↓		
Discrete, aeroshell sequencer 8	↓		½	↓	↓		
Discrete, BVS sequencer 1	↓			↓	↓	½	
Discrete, BVS sequencer 2	↓			↓	↓	↓	
Discrete, BVS sequencer 3	↓			↓	↓	↓	
Discrete, BVS sequencer 4	↓			↓	↓	↓	
Discrete, BVS sequencer 5	↓			↓	↓	↓	
Discrete, BVS sequencer 6	↓			↓	↓	↓	
Discrete, BVS sequencer 7				↓	↓	↓	
Discrete, BVS sequencer 8	Bilevel			½	½	½	
Visual photometer 1 (aeroshell)	0-40 MV			2	2		
Visual photometer 2 (aeroshell)	↓			↓	↓		
Visual photometer 3 (aeroshell)	↓			↓	↓		
UV Photometer 1 (aeroshell)	0-40 MV			↓	↓		
UV Photometer 2 (aeroshell)	0-40 MV			2	2		
Accelerometer, long, high range					20		
Accelerometer lat					20		
Accelerometer vert					20		
Diff press. inflation tank orifice						½	
Press., balloon						↓	1/hr
Temp, balloon						↓	↓
Strain, balloon film 1						↓	↓
Strain, balloon film 2						↓	↓
AGC, receiver	0-5 V					↓	↓
Temp, solar panel 1	↓					↓	↓
Temp, solar panel 2	↓					½	1/hr

TABLE 32.- ENGINEERING AND SCIENCE DATA REQUIREMENTS - Continued

Measurement	Signal level	Format (sps)					
		I	A	B	C	D	E
Temp, solar panel 3	0-5 V					½	1/hr
Temp, solar panel 4							
Temp, solar panel 5							
Temp, solar panel 6	0-5 V						
Current, solar panel 1	0-40 MV						
Current, solar panel 2							
Current, solar panel 3							
Current, solar panel 4							
Current, solar panel 5							
Current, solar panel 6	0-40 MV						
Voltage, sonde 1 battery	0-5 V						
Voltage, sonde 2 battery							
Press., atmospheric, low range							
Press., atmospheric, high range							
Temperature, atmospheric 1							
Temperature, atmospheric 2	0-5 V						
Solar aspect sensor 1	8 bit						
Solar aspect sensor 2	8 bit						
Solar aspect sensor 3	8 bit						
Water vapor detector	0-5 V						
Visual photometer 1 (BVS)							
Visual photometer 2 (BVS)							
Visual photometer 3 (BVS)	0-5 V						
Radar altimeter 1	8 bit						
Radar altimeter 2	4 bit						
Radar altimeter 3	4 bit						
Radar altimeter 4	4 bit						
Radar altimeter 5	4 bit					½	
Mini bio lab	48 bit digital						
Frame sync	21 bits		½	½	½	½	
Frame identity	3 bit		½	½	½	½	1/hr
8 spares	8 x 8 bit		½				
9 spares	9 x 8 bit			½	½		
10 spares	10 x 8 bit					½	
6 spares	6 x 8 bit						1/hr

TABLE 32.- ENGINEERING AND SCIENCE DATA REQUIREMENTS - Concluded

Measurement	Data Rate	Format
Mass spectrometer Emission current Filament current Repeller potential Accelerating potential (min.) Accelerating potential (max.) Electrostatic sector reference potential SEM reference potential Ion pump current	Formatted into one frame (1200 bits) of data once each 6 hr. (20-sec sample time)	F
Gas chromatograph	Formatted into one frame (1200 bits) of data once each 6 hr. (20-minute sample time)	G
Accelerometer, long, low range Accelerometer, X-axis, low range Accelerometer, Y-axis, low range	24 sec of data formatted with sync and time (1200 bit frame) one each 6 hr if preset accelerometer threshold is exceeded.	H
Frequency difference AGC receiver	Formatted with time at 1 sample per minute (240 bit frame every 13 minutes for 1½ hr).	J
Sonde data Pressure, atmospheric Temperature, atmospheric Water vapor Voltage, battery Temp, electronics Frame count	Formatted in sonde (240 bit frame every 4 minutes until impact).	K

TABLE 33.- BVS DATA FORMATS

Format	Period used and description
A	Transmitted during capsule/spacecraft separation. Consists of aeroshell and BVS engineering data.
B	Transmitted during capsule entry (entry minus 5 minutes to BVS separation plus 1 minute). Consists of aeroshell science and engineering and BVS engineering. Also probe status data.
C	Stored during entry and played back after BVS separation. Consists of format B data plus high rate acceleration data.
D	Transmitted during BVS deployment. Consists of BVS atmospheric science and PVS engineering data.
E	Hourly collection format. Consists of atmospheric science, BVS engineering, bio lab, and sonde status.
F	Mass spectrometer data format. Collected once each 6 hr.
G	Gas chromatograph data format. Collected once each 6 hr.
H	Accelerometer format. Consists of a 30-sec period of accelerometer data collected once each 6 hr if the acceleration level exceeds a predetermined threshold.
I	Status monitor data format. Collected during interplanetary cruise.
J	Position determination data format. Spacecraft beacon-BVS receiver oscillator frequency difference measurement collected and stored during each spacecraft pass for later transmission.
K	Sonde data format. Consists of atmospheric science data collected and transmitted by radio sonde. Received and stored in BVS for later transmission.

TABLE 34.- SUBSYSTEM SEQUENCE CAPSULE/SPACECRAFT SEPARATION

T-0 (S-6 min)	Checkout discrete received from aeroshell sequencer
T-0+	Output discretes 1. Data handling subsystem on 2. Transmitter on 3. Transmitter low power discrete 4. Format A discrete 5. Radar on 6. Bio lab on 7. Mass spectrometer on 8. Gas chromatograph on 9. Triad accelerator on
T+01:00	1. Format A off 2. Format B on 3. Format C into storage-on
T+01:20	1. Format B off 2. Format C into storage-off 3. Start playback from storage 4. Triad accel. off
T+01:40	1. Stop playback from storage 2. Format D on
T+02:00	1. Format D off 2. Format E into storage (1 frame)
T+02:10	1. Format F into storage (1 frame) 2. Format G into storage (1 frame)
T+02:30	1. Start playback from storage 2. Radar off 3. Bio lab off 4. Mass spectrometer off 5. Gas chromatograph off
T+05:00	1. Stop playback from storage 2. Format A on
T+11:00	1. Transmitter off 2. Data handling S/S off
T+25:30	1. Data handling S/S on 2. Transmitter on 3. Transmitter low power discrete 4. Format A discrete
T+31:00	1. Transmitter off 2. Data handling S/S off

TABLE 35.- DATA SUBSYSTEM SEQUENCE ENTRY AND DEPLOYMENT

Events	Time	
(E-5)		Entry minus 5 min signal from aeroshell sequencer Output discretes 1. Data handling S/S on 2. Transmitter on 3. Transmitter high power discrete 4. Format B discrete 5. Triad accelerometer on
(E-0)	E-0	(1 g discrete from science instrument) Start Format C into storage
	E+00:30	Stop Format C into storage (A 1 g threshold level after T+00:30 will initiate a 2-sec storage of Format C) 2-sec storage cycle locked out after 10 cycles.
(St-0)	St-0	BVS separation discrete from sequencer
	S_t+01:00	1. Format B stop 2. Start playback from storage 3. Radar on
	St+06:00	1. Stop playback from storage 2. Format D on
(D-0) (PP+00:01:00)	D-0	BVS deployed (signal from sequencer) 1. Bio lab on 2. Mass spectrometer on 3. Arm wind gust measurement (Mode 6)
	D+02:00	1. Format D stop 2. Two frames of Format E into storage
	D+02:10	1. Sample to mass spectrometer 2. One frame of Format F into storage
	D+02:30	1. Start playback of storage 2. Mass spectrometer off
	D+07:30	1. Stop playback of storage 2. Start Format E
	D+15:00	1. Transmitter off 2. Radar off 3. Data handling S/S off
	D+30:00	1. Start bio sample collection

TABLE 35.- DATA SUBSYSTEM SEQUENCE ENTRY AND DEPLOYMENT - Concluded

Events	Time	
	D+1:00:00	1. Data handling S/S on 2. Radar on
	D+1:02:00	1. Two frames of Format E into storage
	D+1:02:10	1. Radar off 2. Data handling S/S off 3. Accelerometer off
	D+2,3,4, and 5	Repeat D+1 to D+1:02:10 sequence
	D+4:30:00	1. Start G.C. sample collection
	D+5:30:00	1. Bio sample to bio lab
	D+5:40:00	1. Gas chromatograph on 2. Sample to G.C. 3. Frame G start
	D+6:00:00	1. Data handling S/S on 2. Radar on 3. Mass spectrometer on 4. Accelerometer on
	D+6:02:00	1. Two frames of Format E into storage
	D+6:02:10	1. One frame of Format F into storage 2. One frame of Format G into storage 3. Wind gust data transferred if threshold was exceeded
	D+6:02:40	1. Radar off 2. Mass spectrometer off 3. Gas chromatograph off 4. Data handling off 5. Arm wind gust measurement
	D+7, 8, 9, 10, and 11 repeat D+1 to D+1:02:10 sequence	
	D+12	Repeat D+6 to D+6:02:40 sequence
		Above sequence continued to end of mission

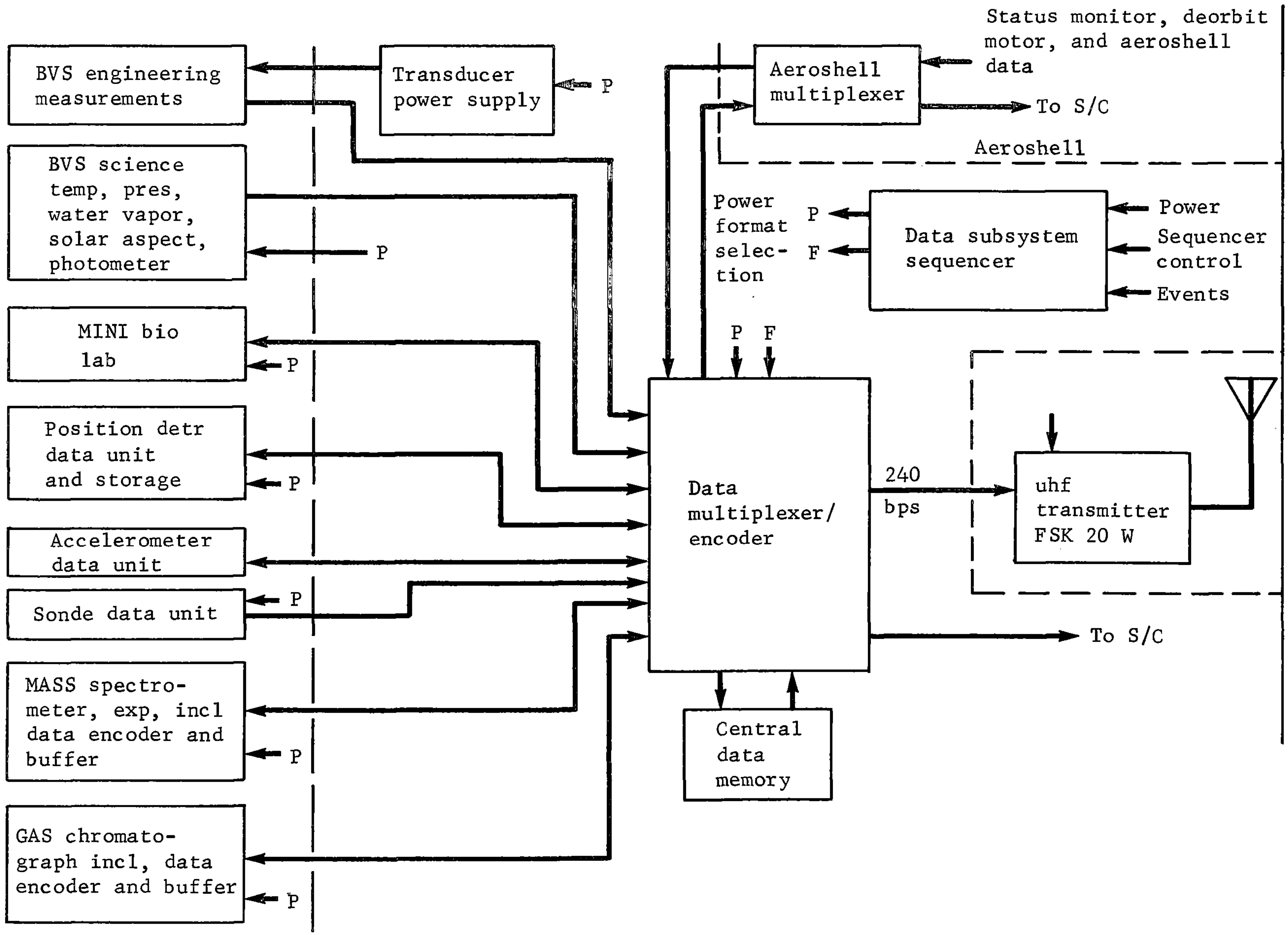


Figure 51.- BVS Data Subsystem Block Diagram

TABLE 36.- DISCRETES ISSUED BY DATA SUBSYSTEM SEQUENCER (DSS)

From DSS		To
1.	Transmitter on/off	BVS power control
2.	Transmitter low power	Transmitter
3.	Transmitter high power	Transmitter
4.	Data multiplexer/encoder, transducer power supply, data memory, atmospheric science instruments on/off	BVS power control
5.	Bio lab on/off	BVS power control
6.	Mass spectrometer on/off	BVS power control
7.	Gas chromatograph	BVS power control
8.	Main receiver on/off	BVS power control
9.	Command detector and decoder on/off	BVS power control
10.	Position determination data unit on/off	BVS power control
11.	Sonde data unit on/off	BVS power control
12.	Arm sonde	BVS power control
13.	Release sonde	BVS power control
14.	Safe sonde	BVS power control
15.	Accelerometer on/off	BVS power control
16.	Radar on/off	BVS power control
17.	thru 25. Formats A-E start/stop	Data multiplexer/ encoder
26.	Atmospheric sample collection	Science
27.	Bio sample collection	Science
28.	thru 40. Science discretes	Science
41.	thru 50. Data handling discretes	Data multiplexer/ encoder

The data memory is sized by the amount of data collected during the time it takes the spacecraft to make one orbit. This results in a storage of approximately 36 000 bits. This memory also provides storage of entry data for the period during which transmission may be interrupted due to rf blackout.

The transducer power supply provides 5 Vdc regulated power to those engineering sensors requiring regulated power.

Size, weight, and power required for the data subsystem are shown in table 37.

TABLE 37.- DATA SUBSYSTEM SIZE, WEIGHT, AND POWER

Component	Size, cu in.	Weight, lb	Power, W
BVS			
Data subsystem sequencer	280	4.0	4.0
Data multiplexer/encoder	420	10.0	8.5
Memory	200	4.0	1.0
Transducer power supply	20	0.5	1.0
Engineering sensors	20	0.5	----
Total	940	19.0	14.5
Aeroshell			
Multiplexer/encoder	280	3.0	4.0
Transducer power supply	20	0.5	1.0
Engineering sensors	80	2.0	----
Total	380	5.5	5.0

Subsystem operation: The data subsystem provides for the collection, processing, storage, and transmission of science and engineering data from the BVS during the planetary mission. In addition, it provides the primary source of system checkout data during prelaunch tests. During interplanetary cruise, capsule status data are multiplexed, converted, and formatted by the aeroshell multiplexer under control of a format generator in the aeroshell. Clocking and data address are provided by the spacecraft. After attaining orbit around Venus and on Earth command, the spacecraft issues a separation discrete to the aeroshell sequencer.

This sequencer controls the capsule separation and entry sequence. The data subsystem sequencer in the BVS is turned on by the aeroshell sequencer and receives a discrete that initiates the data handling sequence for separation. This sequence is shown in table 34. The aeroshell multiplexer is clocked and addressed by the data multiplexer/encoder in the BVS. All aeroshell data and deorbit engine data are multiplexed by the aeroshell multiplexer and formatted by the data multiplexer/encoder in the BVS. This results in a simpler interface between the aeroshell and BVS and also reduces the amount of hardware in the BVS.

After deorbit engine separation, the data subsystem is shut down until 5 minutes before entry when the aeroshell sequencer again turns on the DSS and issues a discrete to initiate the data handling sequence for entry and deployment. This sequence is shown in table 35. Certain functions are event-controlled while others are timed. Those that are event-controlled are turn on for entry, start storage of entry data (initiated by g switch), playback of entry data (timed from BVS separation signal), and BVS-deployed discrete. All data collection sequences after deployment are timed from deployment. Transmission of data during the second and succeeding passes of the spacecraft is also event-controlled and results from a received signal from the spacecraft exceeding a predetermined threshold. This sequence is shown in table 38. The data subsystem will continue to operate in an automatic mode until end of mission.

Mechanization options: The major tradeoff considered in the data subsystem is in the use of a decentralized versus a centralized data subsystem. A centralized system requires a small general purpose computer to provide flexibility for handling various experiments. All data processing, data compression, and formatting would be handled through this computer. In the decentralized approach, the data processing for each unique experiment is handled separately. A standard digital interface is established between a central data multiplexer and the data processing electronics for each unique experiment. Because of the possible experiment complement that may be flown, considerable flexibility must be provided to use a centralized approach. A number of experiments identified in the baseline configuration have unique data processing requirements. Reception, detection, and storage sonde data will occur only twice during the mission. Better use of power is possible with separate electronics for this experiment and turning it on for only the periods required.

TABLE 38.- TRANSMISSION SEQUENCE AFTER BVS IS DEPLOYED

D+20:00:00; etc (at orbit period intervals)	Receiver on
D+24+(R-0)	Spacecraft beacon received 1. Transmitter on 2. Command decoder on 3. Position determination data unit on 4. Data handling S/S on (240 bps bit rate) 5. Radar on 6. Science instruments on 7. If in data collection cycle, complete cycle and initiate Format E; if not in cycle, lock out collection cycle and initiate Format E.
R+2:30(+0 to +00:40) (02:40 for collection cycle)	1. Format E stop 2. Start playback of central storage (includes doppler data) (36 000 bits total)
R+05:00	1. Stop playback of central storage 2. Start Format E
R+05:30	1. Stop Format E 2. Start sonde data storage playback (or if no sonde dropped initiate second playback of central storage
R+08:00 (or +09:00 if sonde has been dropped)	1. Stop playback from central storage 2. Start Format E
R+09:00 (or +10:00)	1. Stop Format E 2. Unlock data collection cycle 3. Data handling S/S off 4. Transmitter off 5. Command decoder off 6. Radar off 7. Science instruments off
R+01:30:00 (or loss of beacon plus 1 min)	1. Position determination data unit off 2. Receiver off

Other experiments that have unique data processing requirements are the mass spectrometer, gas chromatograph, and wind gust detection. Wind gust detection requires a high sampling of an accelerometer when the input exceeds some predetermined threshold. The gas chromatograph requires a comparatively long time (20 minutes) to process one sample and the data sampling rate is low. Data compression will be used on the mass spectrometer data to reduce the total transmitted (and stored) data. This will involve the detection and storage of peak signals with the data system being sized to handle 48 peaks. It results in a compression of approximately 10 to 1 for these data.

A decentralized system has been selected for the BVS mission. It results in the use of a standard digital interface between a central data multiplexer and unique experiment data processors. It will also result in a minimum weight system because experiment data processors are added only with the experiment. With a centralized design the capability would be included in the basic design and could not readily be removed if the experiment did not fly.

Telecommunications for the Venus Flyby Mission

Differences between the BVS/entry vehicle telecommunications for the flyby and the orbital mission are identified in this section.

Radio subsystem.- Requirements for the BVS/entry vehicle radio system are shown in table 39. Note that the requirements are identical to those of the orbiter-supported BVS up to and including the BVS deployment phase. Following that event the relay communications link to the S/C is terminated and a direct-to-Earth link is established for telemetry and command operations.

The BVS radio subsystem consists of a 390 MHz, 20 W output, frequency shift keyed relay transmitter, and 390 MHz relay antenna plus an S-band receiver, modulator/exciter, power amplifier, diplexer, and antenna as shown in the block diagram of figure 52.

TABLE 39.- RADIO SUBSYSTEM REQUIREMENTS

Preentry through station deploy mission phases
Provide a data transmission link to the supporting spacecraft during separation from the spacecraft, ΔV boost, and entry through deploy of the BVS. (An entry blackout of communications for approximately 20 sec is acceptable.)
Provide a minimum data rate capability of 240 bps at a maximum bit error rate probability of 4×10^{-3}.
Buoyant station postdeploy mission phases
Provide a telemetry data transmission link to Earth having a minimum bit rate capability of 30 bps at a bit error rate of 5×10^{-3}.
Provide a receiving capability for Earth-originated commands and deliver a composite sync and data signal to the command subsystem.
Provide a transmitter frequency that is coherent with the received frequency.
Provide a turnaround ranging channel.

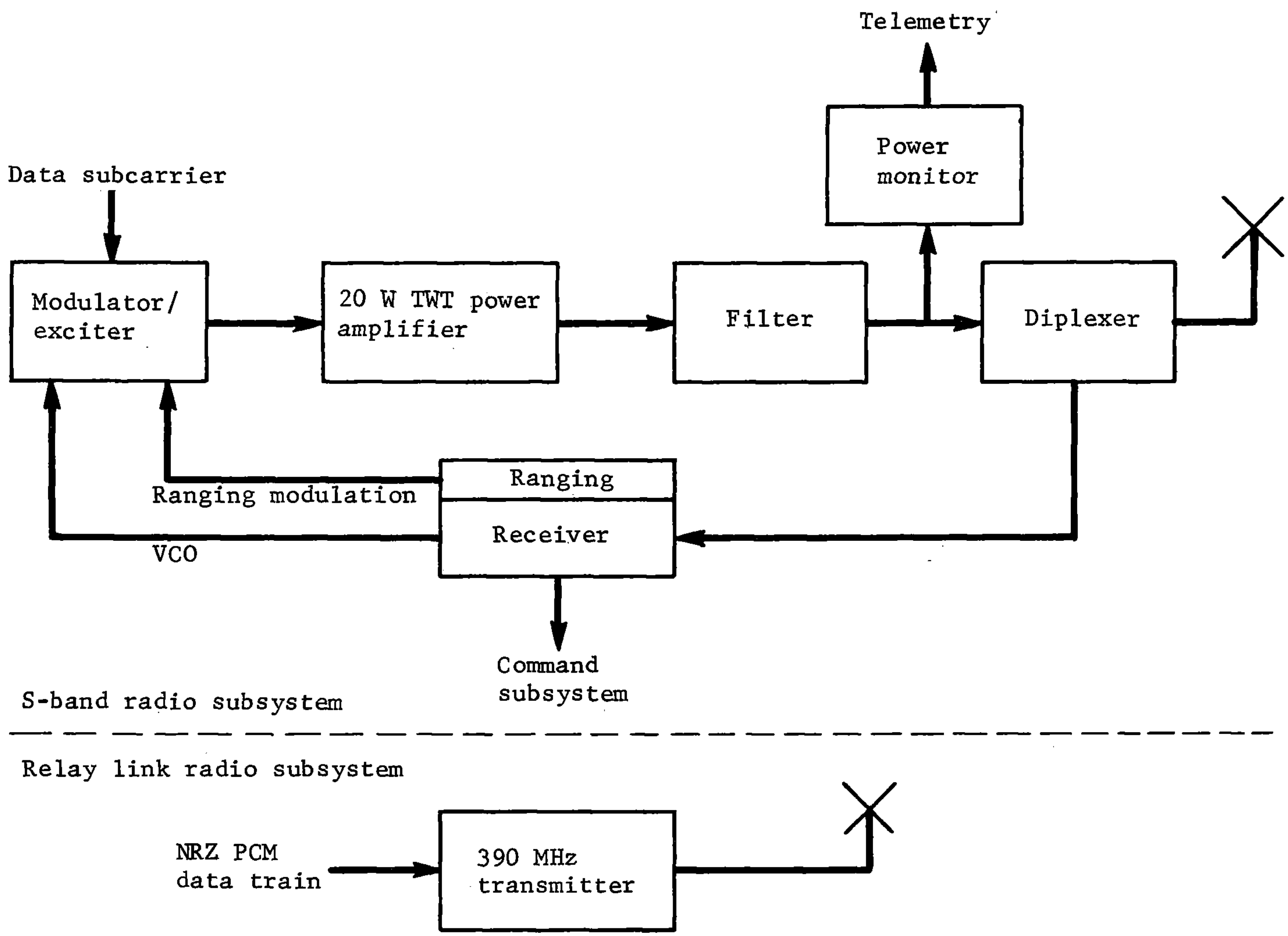


Figure 52.- BVS Radio Subsystem

The relay transmitter and antenna are identical to the ones used for the orbiter-supported mission described in the previous section. Both the relay transmitter and antenna are separated from the BVS when the deployment phase is completed to reduce the floated weight of the station.

The S-band receiver is a narrow band, double superheterodyne unit with automatic phase control (APC) and automatic frequency search and acquisition features. When the APC loop is locked to the signal transmitted from Earth the receiver controls the phase and frequency of the S-band carrier being transmitted to Earth and also demodulates the command signal. A correlation detector in the receiver detects lock loop (signal acquisition), calibrates the signal strength, and provides automatic gain control (AGC). The lock loop signal causes the exciter drive to be derived from the receiver's VCO. This feature provides a two-way coherent doppler capability. When the loop is unlocked the frequency is derived from a fixed crystal-controlled oscillator in the modulator/exciter. For turnaround ranging a quadrature detector detects the wide band ranging signal received from Earth, and upon command the signal is used to modulate the return link.

Receiver characteristics are shown in table 40. Performance is essentially that of the Mariner 1969 unit with a frequency search mode added. The APC bandwidth is increased from the 20 Hz of the Mariner unit to 70 Hz to provide a search and acquisition time of approximately 87 sec to lock onto the command carrier.

TABLE 40.- S-BAND RECEIVER CHARACTERISTICS

Parameter	Value
Type	Double superheterodyne with APC and frequency acquisition sweep
Frequency	2115 ± 5 MHz
Noise figure	8 dB
Carrier tracking loop	$2\ B_{LO}$ = 70 Hz, S/N = 6 dB
Predetection bandwidth	4.5 kc
Dynamic range	-150 to -70 dBm
Weight	5 lb
Power	2.5 W
Size	150 cu in.
Others	Ranging channel

A modulator/exciter having characteristics shown in table 41 is phase modulated by a single square wave subcarrier that has been biphase modulated by 30 bps PCM telemetry data. The output is used to drive a 20 W traveling wave tube (TWT) power amplifier. The power amplifier and integral power supply characteristics are shown in table 42. They are based on Mariner 1969 performance. Some improvement in efficiency can be expected for models to be used for a 1972 mission but the table is representative of minimum performance anticipated.

TABLE 41.- MODULATOR/EXCITER CHARACTERISTICS

Parameter	Value	
Frequency	2295 ± 5 MHz	
Power output	60 MW	
Phase stability	6° peak, auxiliary mode 12° peak, coherent mode	
Frequency stability		
Short term	10^{-7} (20 min)	Auxiliary mode
Long term	10^{-6} (12 hr)	Auxiliary mode
Modulation bandwidth	1.8 MHz	
Peak modulation	±2 rad	
Modulation sensitivity	1 rad ± 5% per V peak	
RF input impedance	50 ohms VSWR less than 1.2:1	
RF output impedance	50 ohms VSWR less than 1.4:1	
Power	2 W	
Weight	3.0 lb	
Size	90 cu in.	

A diplexer is required to permit sharing of a common transmit/receiver antenna. Isolation of 80 dB is provided. Other characteristics are shown in table 43.

A typical S-band antenna for the direct link is a cavity-backed, crossed-slot type with a helical feed and a halfpower beamwidth of 130° (ref. 7). A sketch and list of characteristics are shown in figure 53.

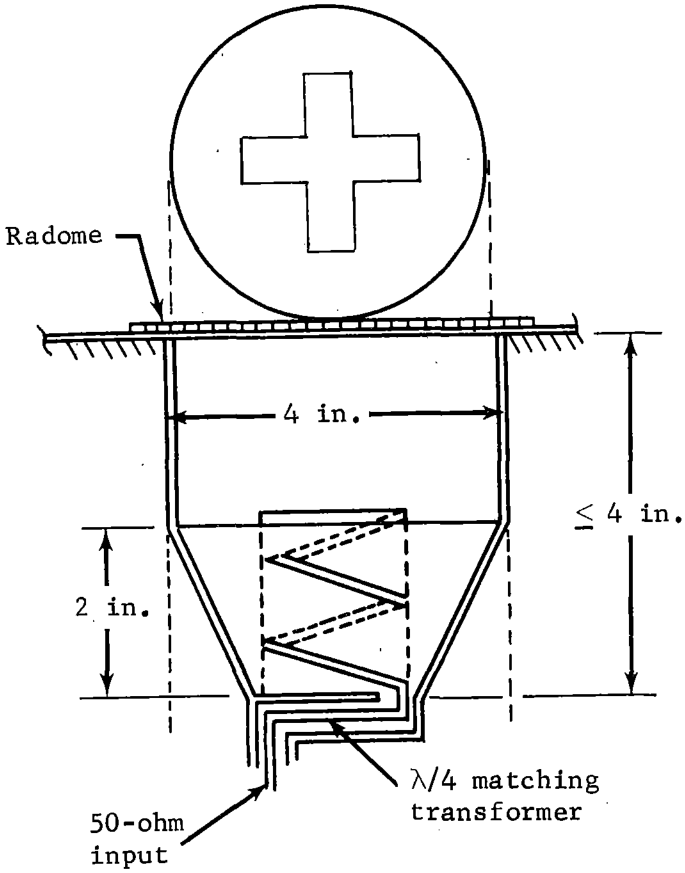
A summary list of BVS radio subsystem equipment with estimates of size, weight, power, and development status is shown in table 44.

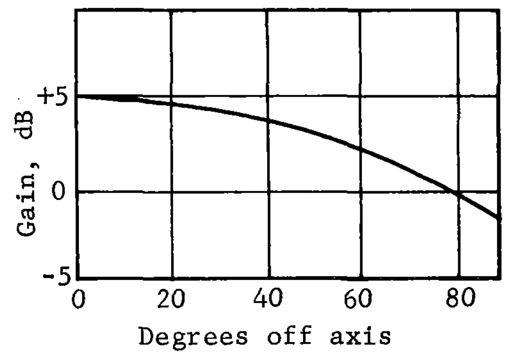
TABLE 42.- POWER AMPLIFIER AND POWER SUPPLY CHARACTERISTICS

Parameter	Value
Type	TWT
Frequency	2295 ± 5 MHz
Power output	20 W
RF gain saturation	35 dB
Noise figure	35 dB
RF impedance	50 ohms
VSWR	1.2:1 maximum
Turn on time	90 sec
Spurious output	Less than -80 dBm in any 1 Hz band in the 2110 to 2120 MHz range
Harmonics	60 dB below unmodulated carrier
Weight	7.8 lb
Power	84 W
Size	200 cu in.

TABLE 43.- DIPLEXER CHARACTERISTICS

Parameter	Value
Frequency	
Transmit	2295 ± 5 MHz
Receiver	2115 ± 5 MHz
Passband insertion loss . . .	0.36 dB maximum, transmit and receive
Impedance	50 ohms
Passband VSWR	1.2:1 maximum
Isolation	80 dB minimum (both frequencies)
Power handling capability . .	100 W
Weight	0.5 lb





Parameter	Value
Peak gain	5 ± 0.5 dB
Half power beamwidth. . .	130°
Polarization	RH circular
VSWR.	1.2:1
Axial ratio	1.5 dB, on-axis
Weight	0.6 lb
Size	4 in. high x 4 in. diam

Figure 53.- BVS S-Band Antenna Characteristics

TABLE 44.- RADIO SUBSYSTEM SIZE, WEIGHT, AND POWER

Component	Weight, lb	Power, W	Size, cu in.	Development status[a]
Transmitter, 390 MHz, 20 W	4.0	67	96	SD, SOA
Relay antenna	5.0	----	895	SD, SOA
S-band receiver	5.0	2.5	150	SD, SOA
Modulator/exciter	3.0	2.0	90	MI
S-band power amplifier (including rf filter and power supply)	7.8	84.0	200	MI
S-band diplexer	0.5	----	50	SD, SOA
S-band antenna	0.6	----	4 in. diam x 4 in. high	SD, SOA

[a] SD = special design to performance and environment.
SOA = conforms to present state of art.
MI = Like Mariner 1969, but improved packaging for environment.

Other units having radio subsystems, such as the subsonic probe and radio drop sondes, are considered to be science experiments and are identical to those used for the orbiter supported BVS. They will not be described here.

The performance of the BVS relay link and the subsonic probe relay link, the latter of which is activated after separation of the BVS from the entry vehicle, are identical, as discussed later.

Relay communication link geometry: A summary of the BVS-spacecraft communications geometry is shown in figures 54 thru 56. Variations in BVS antenna aspect angle, α_p, spacecraft antenna aspect angle, α_B, and communication range, are shown for lead times of 40, 45, and 50 minutes. The angle convention is shown in figure 57.

An entry flight path angle, γ_E, of -30°, targeting parameter, β, of 43.5°, and a spacecraft lead time of 45 minutes were selected for the nominal entry conditions. Selection of these entry conditions resulted from parametric studies performed to optimize entry parameters such as deorbit velocity, coast time from spacecraft separation to BVS entry, angle of attack considerations, etc. These nominal entry conditions provide at least 55 minutes minimum view time. This is the estimated descent time from subsonic probe deployment to impact.

Minimum communication range and nominal spacecraft antenna aspect angles occur near entry, exhibiting little change until after BVS/entry vehicle operation. The BVS antenna aspect angle experiences its maximum change during this period, resulting in increased BVS antenna pointing loss. The increased pointing loss coupled with near maximum communications range results in a worst-case system performance margin occurring near BVS/entry vehicle separation.

Variations in the spacecraft antenna aspect angle establish the required spacecraft antenna beamwidth. Figures 55 and 58 show the aspect angle variations and the assumed spacecraft antenna beamwidth, respectively. A fixed spacecraft antenna boresight was selected to provide adequate coverage for spacecraft lead times of 40 to 50 minutes. These lead times provide view times of 36.6 to 46.6 minutes, respectively. The selection of a spacecraft antenna boresight for a specific lead time would result in improved margins at BVS entry.

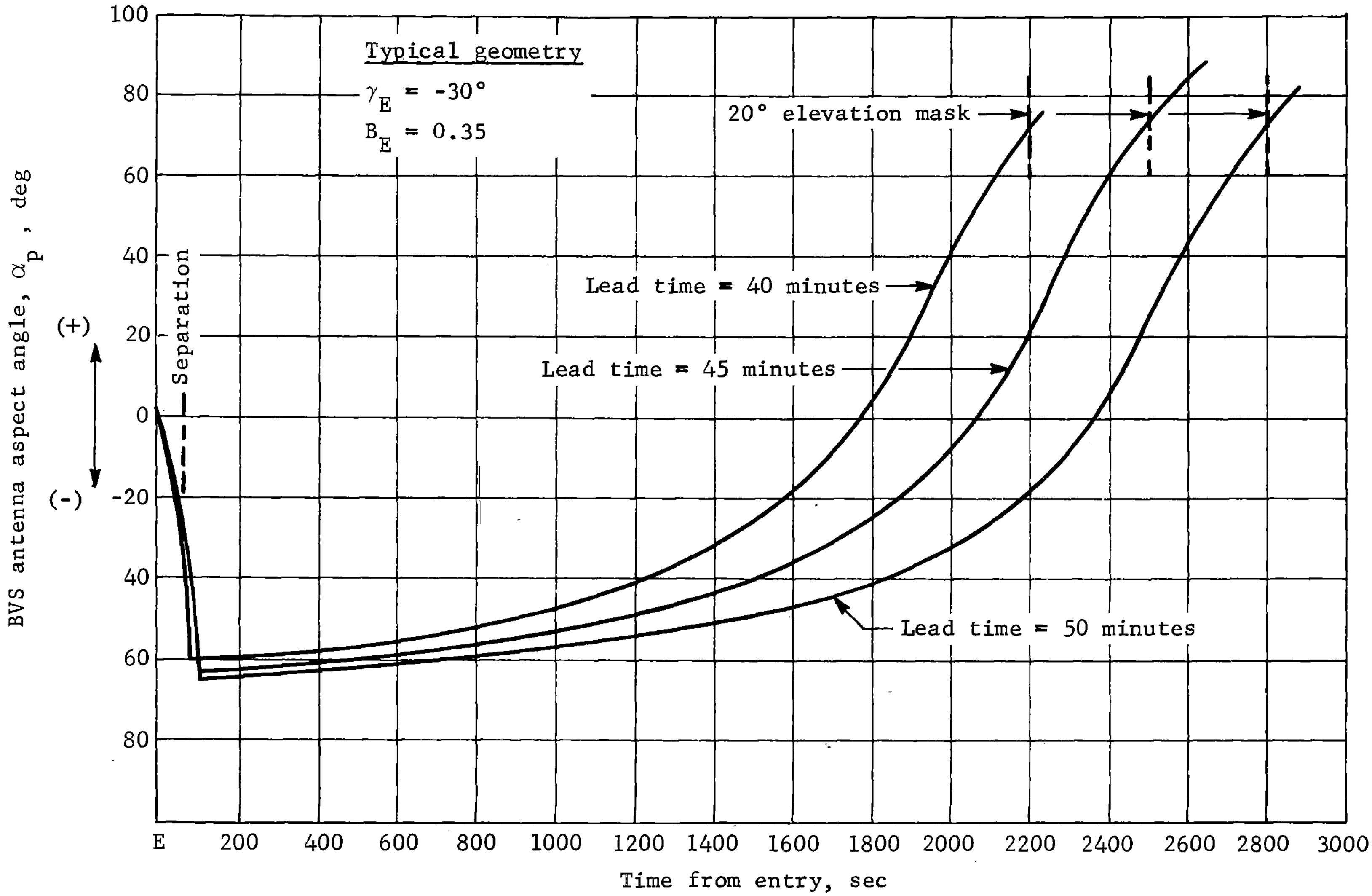


Figure 54.- BVS Antenna Aspect Angle

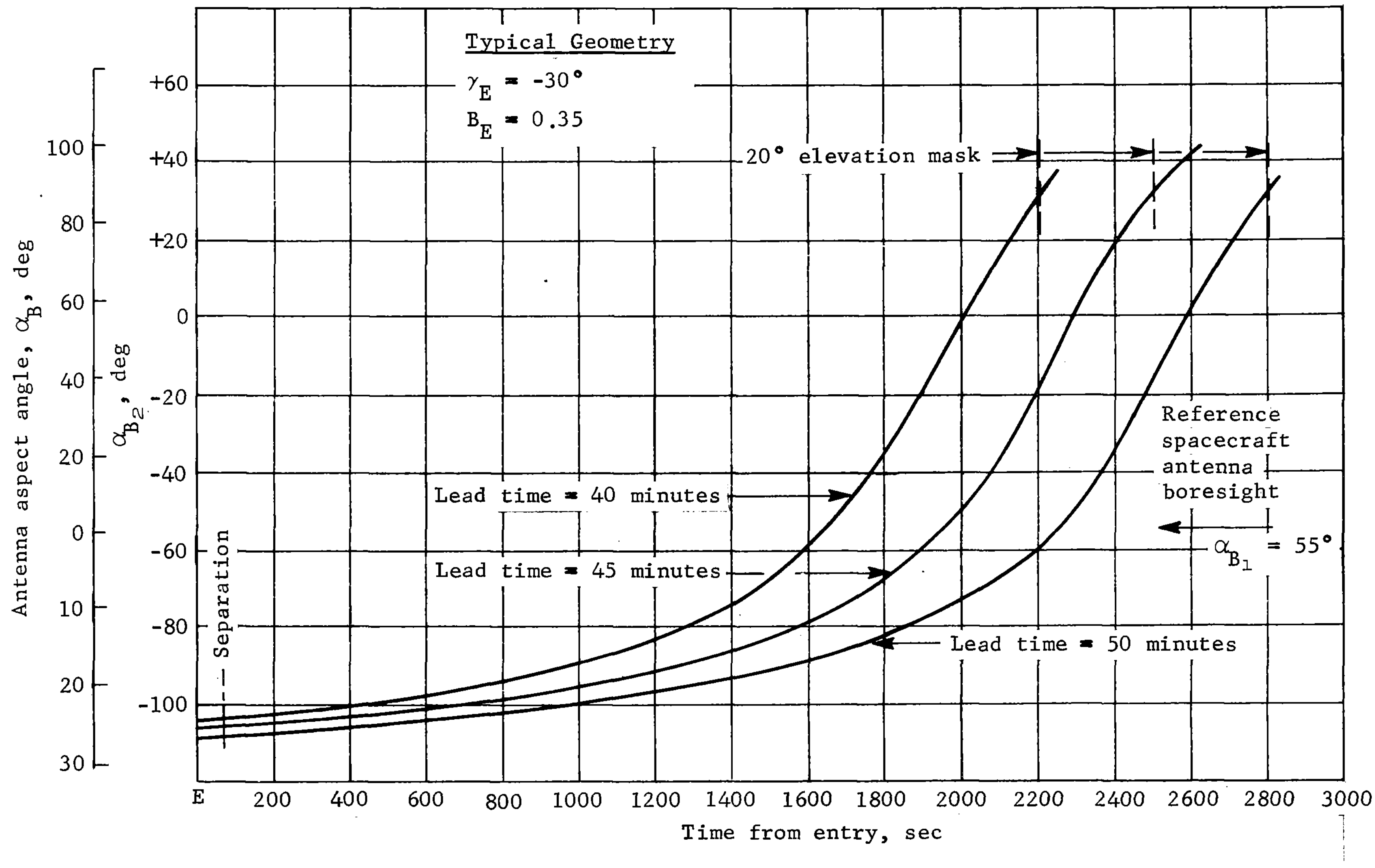


Figure 55.- Spacecraft Antenna Aspect Angles

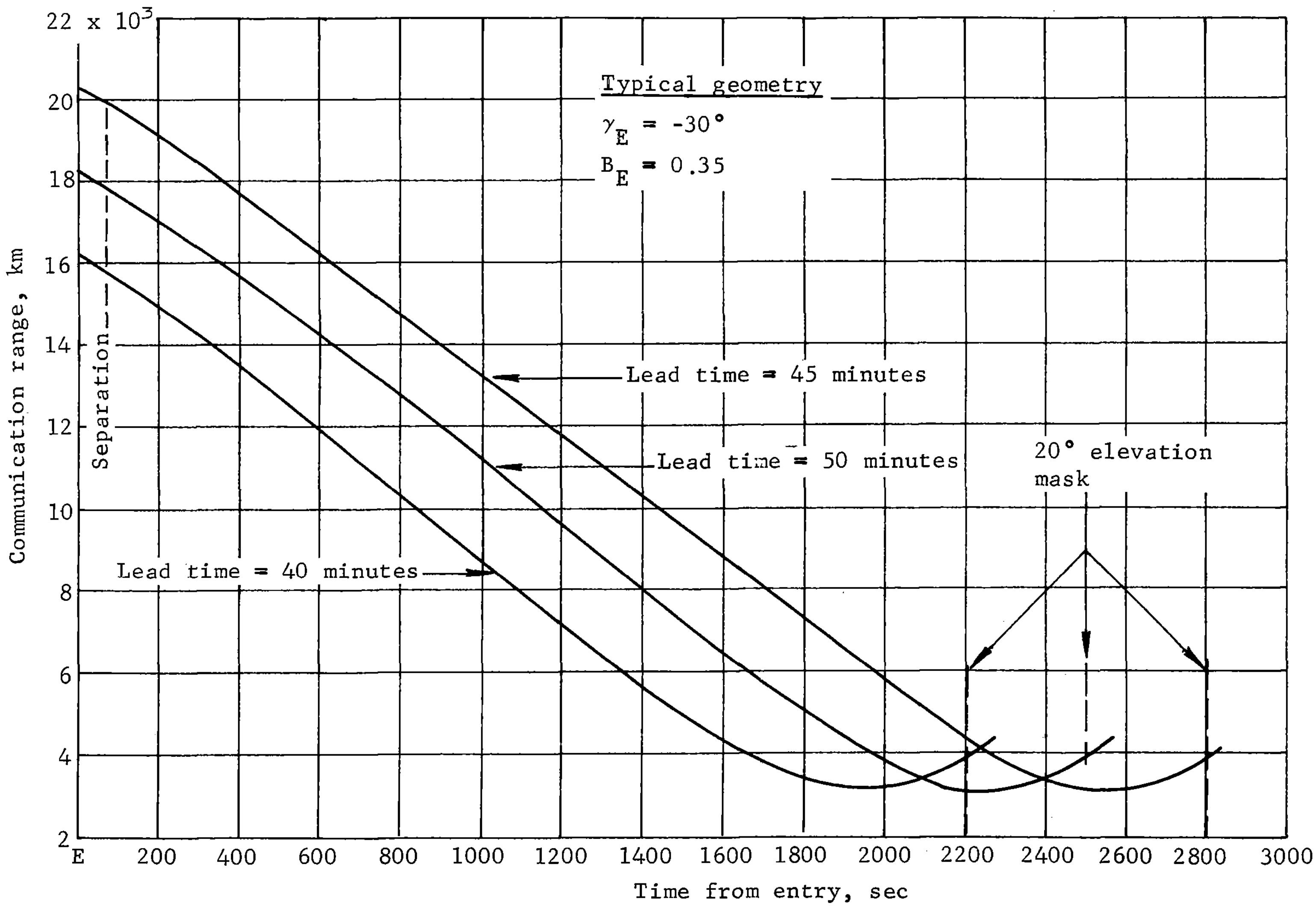


Figure 56.- Communications Range (Relay Link)

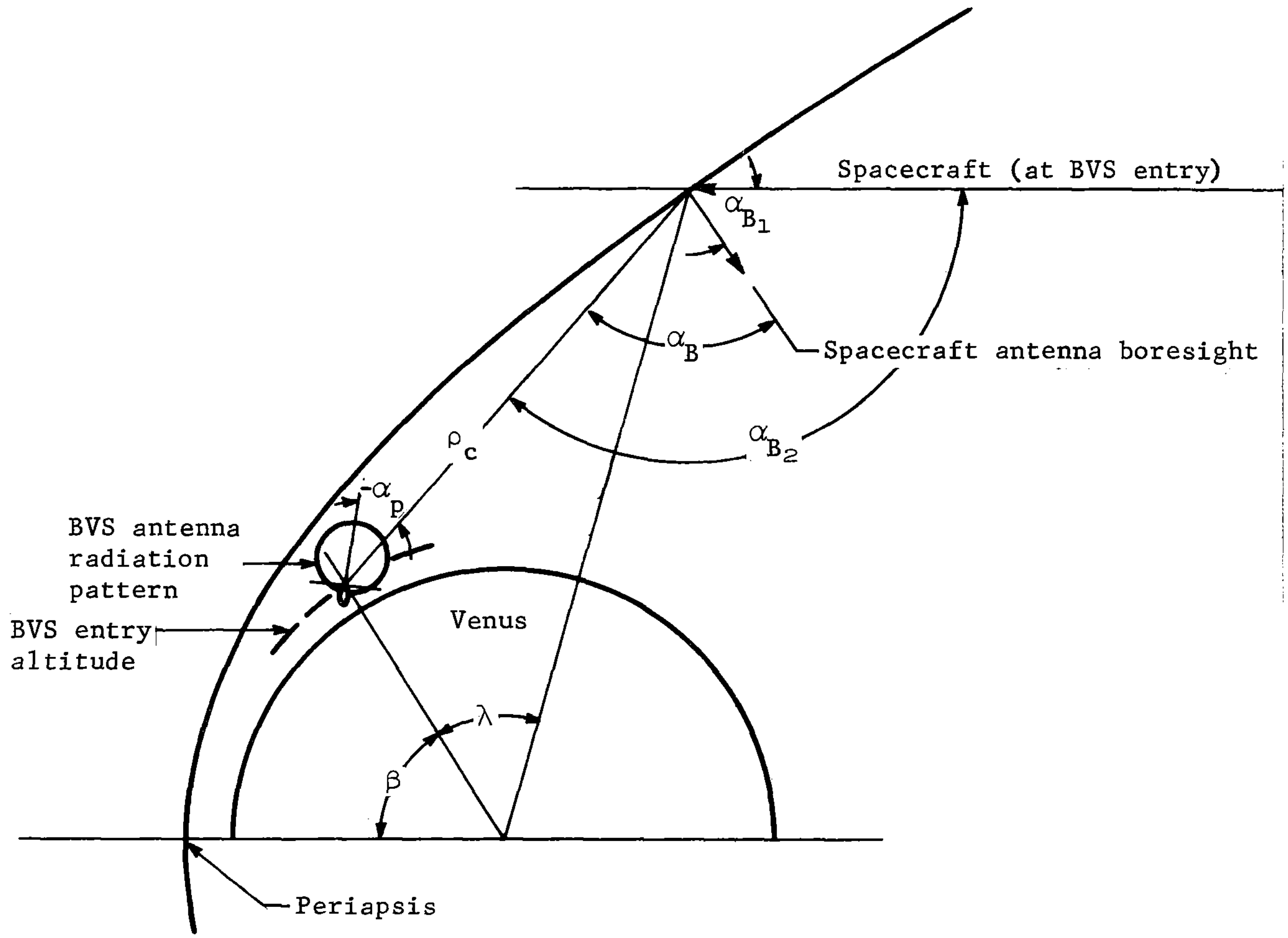
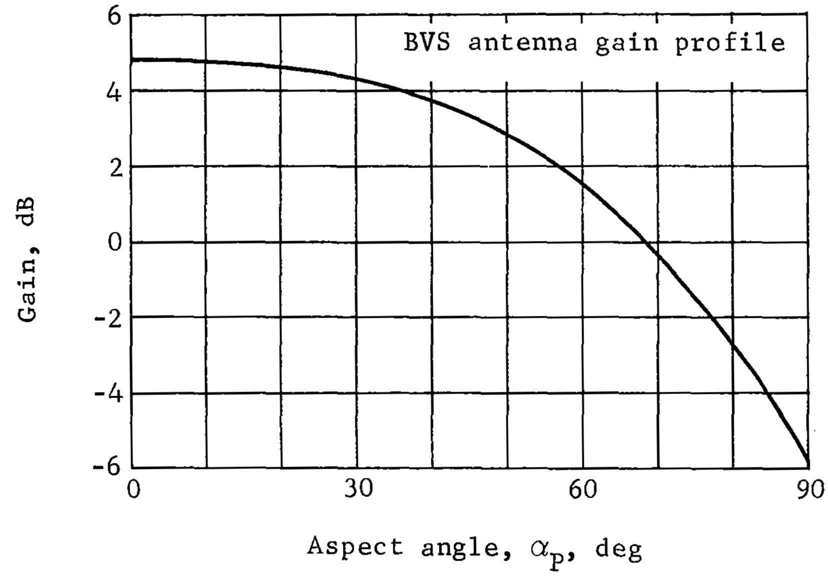


Figure 57.- Entry Geometry Convention



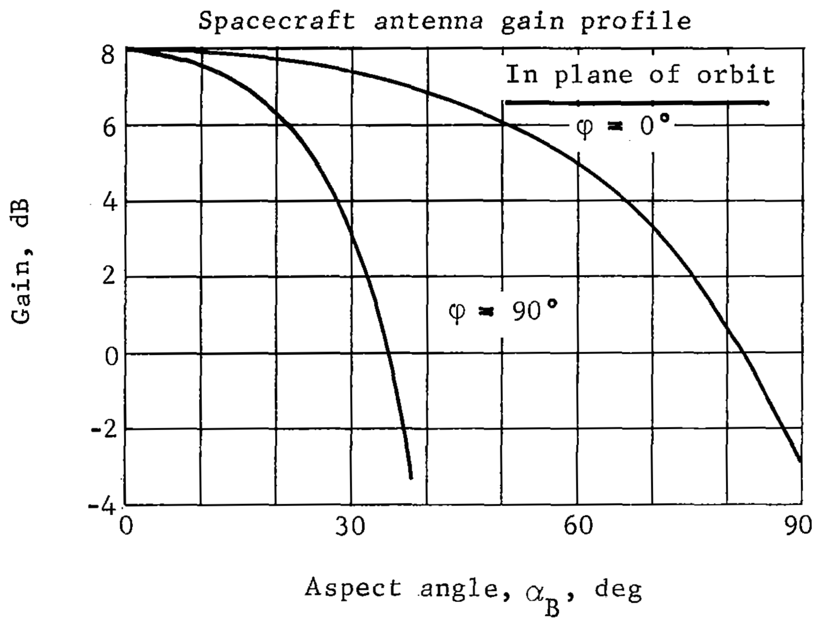


Figure 58.- Spacecraft and BVS/Subsonic Probe Relay Link Antennas (Circular Polarization Pattern)

Relay link performance: The system performance was evaluated for the two extreme atmosphere models. The only significant effect noted on the relay link performance was the difference in time from BVS entry to deployment and from probe release to impact, as noted in the flight mechanics section. The geometry was essentially the same for both atmosphere models, resulting in comparable system performance.

The system performance margin will vary with spacecraft lead time as shown in figure 59. The performance margin shown was derived for the nominal entry conditions noted. An improvement in performance margin, from that shown at entry, will occur for the 20-minute time period before entry, since the communication range will be shorter and comparable antenna aspect angles will exist. The sample design control table (table 45) shows performance margin calculations for point A on figure 59.

The BVS and subsonic probe relay communication systems are identical from the standpoint of effective radiated power, information bit rate, and modulation. Therefore, a single profile can be used to represent the performance of both systems. As for the orbital mission, the fact that the BVS will float at approximately 58 km altitude, while the subsonic probe descends to the surface, will not affect the geometry significantly because the communication range to the spacecraft is much greater than the difference between the BVS and subsonic probe altitudes.

The fading margin required to overcome the slow fading environment is shown on the performance margin profiles of figure 59. Methods used in determining fading margins are the same as for the BVS mission.

A 20-sec communication blackout period is expected to occur approximately 5 sec after BVS entry. Data collected during the blackout period will be recovered through delay transmission techniques in the BVS telecommunications subsystem.

Constraints imposed on the communication system and their effect on system performance are shown in figure 60. The targeting parameter, β, is constrained to upper and lower limits by maximum flight path angle and a 20° elevation mask, respectively. The spacecraft lead time is constrained on the left by the required 35-minute view time and on the right by system performance (communication range). Compliance with these system constraints require that the entry parameters be selected from within the constraint lines.

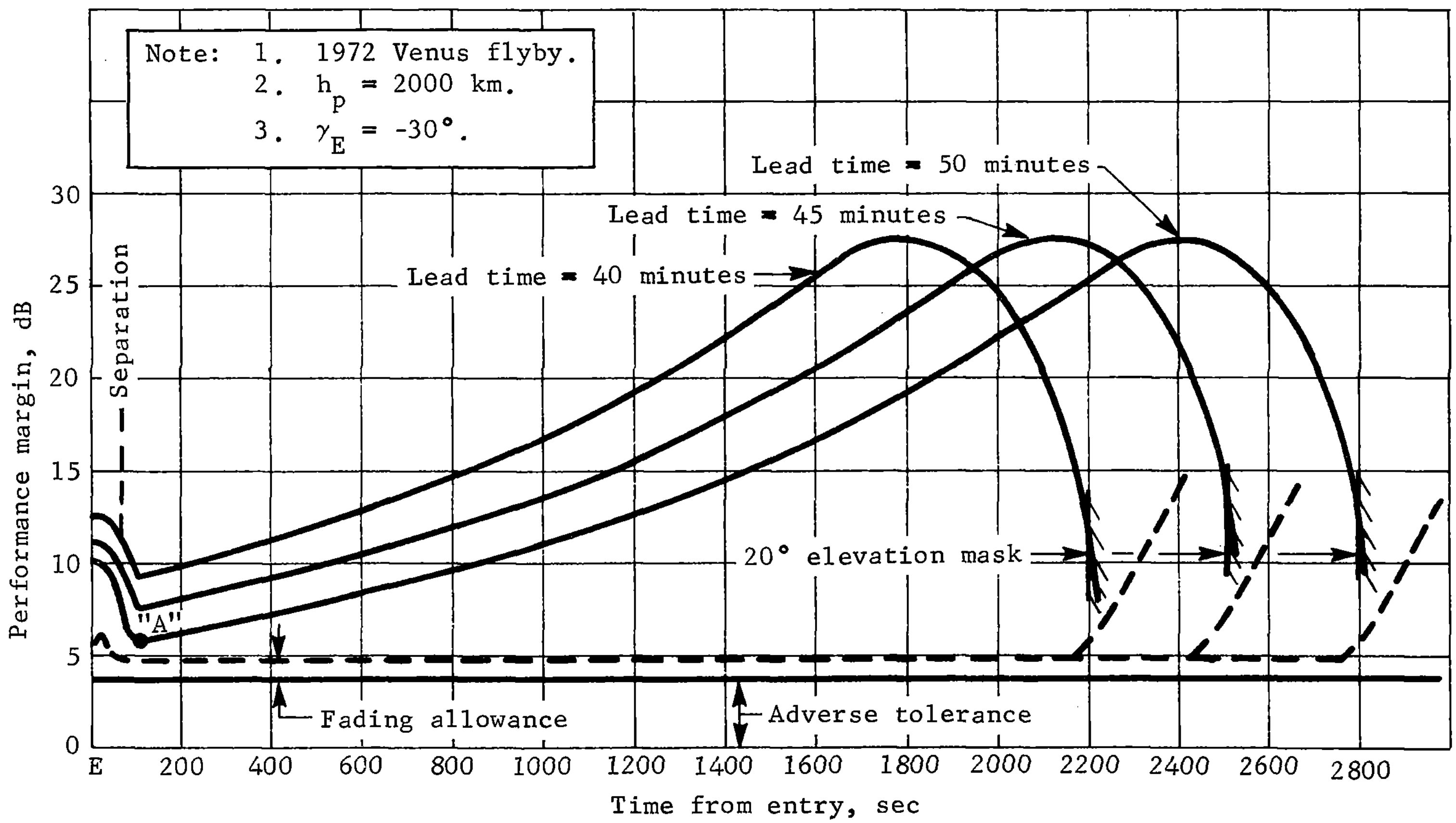


Figure 59.- Relay Communications Performance Margin

TABLE 45.- 1972 ENTRY FROM FLYBY RELAY LINK PERFORMANCE, TELECOMMUNICATIONS DESIGN CONTROL TABLE

Parameter	Value	Tolerance		Notes
		+	-	
Transmitter power	+43.0 dBm	0.2	0.0	20 W min
Transmitting circuit loss	-1.0 dB	0.2	0.3	
Transmitting antenna gain	+4.8 dB	0.5	0.3	
Transmitting antenna pointing loss	-4.3 dB	----	----	-65°
Space loss, F = 390 MHz, ρ_c = 19 780 km	-170.4 dB	----	----	
Polarization loss	-0.3 dB	0.2	0.2	
Ionosphere attenuation	-0.7 dB	0.6	0.7	
Receiving antenna gain	+8.0 dB	0.5	0.5	
Receiving antenna pointing loss	-2.1 dB			-52°
Receiving circuit loss	-0.5 dB	0.0	0.7	
Net circuit loss	-166.5 dB	2.0	2.7	
Received carrier power	-123.5 dBm	2.2	2.7	
Receiver noise spectral density, T system = 786°K	-169.6 dBm/Hz	1.0	1.0	T(max) = 1000°K
Predetection noise bandwidth, bit rate = 240 bps	+44.3 dB-Hz	----	----	27 kHz
Receiver noise power	-125.3 dBm	1.0	1.0	
Carrier-to-noise ratio	+1.8 dB	3.2	3.7	
Threshold carrier-to-noise ratio, $P_e^b = 4 \times 10^{-3}$, TW = 112	-3.9 dB			
Fading allowance	+1.2 dB	----	----	
Performance margin	+4.5 dB	----	3.7	
Worst-case margin	+0.8 dB	3.2	----	

Note: Calculations for point A of figure 59.

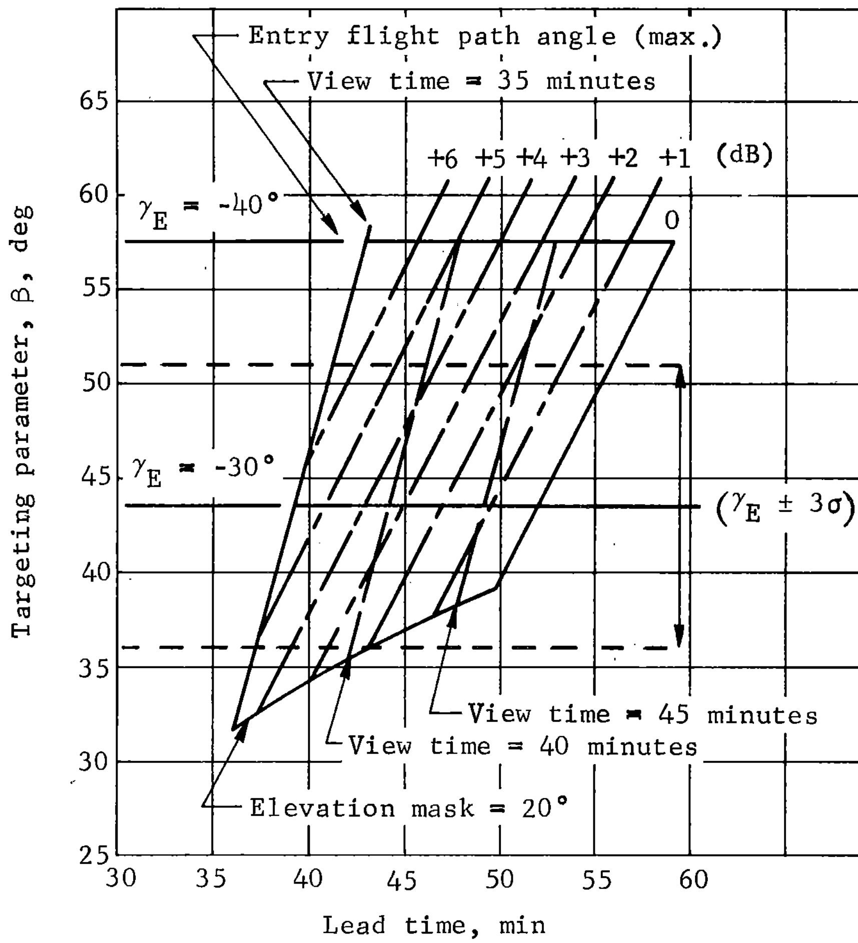


Figure 60.- Constraints of the Relay Communications Link

The system performance constraint was derived for conditions which exist at the lowest system margin point in the deorbit trajectory (near BVS/entry vehicle separation). The 0 dB constraint line is associated with a communication range and antenna aspect angles that result in a system performance margin equal to the system adverse tolerance (including fading allowance). The lines approximately parallel to the 0 dB constraint line represent those entry conditions that result in a worst-case performance margin of +1 dB to +6 dB above the adverse tolerance level. The worst-case system performance margin is a function of the BVS antenna aspect angle and the communication range if the spacecraft antenna is boresighted to provide a constant angle to the BVS, at entry, over the range of flight path angles considered. The constant margin lines were developed by determining entry conditions that result in constant BVS antenna aspect angles and communication range at the worst-case performance margin point. Entry conditions that provide a minimum view time of 40 and 45 minutes are identified in figure 60.

S-band telemetry performance: Following the BVS deployment phase the S-band system is switched into operation. When the command receiver completes its acquisition search, and detects lock to the carrier being transmitted from Earth, the S-band transmitter is activated and operates at a frequency that is coherent with the received frequency.

The Earth-to-BVS link analysis is given in the section following that discusses the command subsystem. The BVS-to-Earth telemetry link parameters are shown in table 46. Note that the margin is adequate for the conditions shown, because it equals or exceeds the sum of the adverse tolerances.

A range of 93 x 10^6 km was chosen as maximum. This corresponds approximately to the range from Venus to Earth 15 days after station deployment. Since range increases with time for the period of interest, this should represent the worst case.

Antenna pointing loss is shown as 3.5 dB which represents an antenna aspect angle to Earth of 70° (20° above horizontal). This represents the desired limit. The initial aspect angle to Earth at station deployment is 46° from vertical with a predicted wind drift that should reduce the angle (improve the effective gain) as the station moves toward the terminator.

TABLE 46.- 1972 TYPE I DIRECT LINK PERFORMANCE, TELECOMMUNICATION DESIGN CONTROL TABLE

Parameter	Value	Tolerance +	Tolerance -	Notes
Total transmitter power	+43.0 dBm	1.0	0.0	20 W minimum
Transmitting circuit loss	-1.0 dB	0.5	0.5	
Transmitting antenna gain	+5.0 dB	0.5	0.5	
Transmitting antenna pointing loss	-3.5 dB	3.5	0.0	±70°
Space loss, F = 2295 MHz, ρ_c = 93 x 10^6 km (0.62 A.U.)	-258.3 dB	----	----	
Polarization loss, ϵ = 6 dB, DSIF ϵ = 0.8 dB	-0.6 dB	----	----	Maximum
Receiving antenna gain	+61.0 dB	1.0	1.0	EPD-283
Receiving antenna pointing loss	-0.3 dB	0.3	0.0	±0.02°
Receiving circuit loss	-0.2 dB	0.1	0.1	Estimate
Net circuit loss	-197.9 dB	5.9	2.1	
Receiver noise spectral density, T system = 45 ± 10°K	-182.1 dBm/Hz	1.1	0.9	EPD-283
Carrier modulation loss	-6.0 dB	0.4	0.4	θ = 1.05 rad ± 5%
Received carrier power	-160.9 dBm	7.3	2.5	
Carrier APC noise BW, 2 B_{LO} = 12 Hz	+10.8 dB	0.5	0.0	MC-4-310A
<u>Carrier performance - telemetry</u>				
Threshold SNR in 2 B_{LO}	+6.0 dB	0.5	1.0	MC-4-310A
Threshold carrier power	-165.3 dB	2.1	1.9	
Performance margin	+4.4 dB	9.4	4.4	
<u>Data channel</u>				
Modulation loss	-1.25 dB	0.4	0.4	θ = 1.05 rad ± 5%
Received data subcarrier power	-156.1 dBm	7.3	2.5	
Bit rate (1/T = 30 bps)	+14.8 dB	----	----	
Required ST/N/B	+6.8 dB	1.0	1.0	$P_e^b = 5 \times 10^{-3}$
Threshold subcarrier power	-160.5 dBm	2.1	1.9	
Performance margin	+4.4 dB	9.4	4.4	

Radio subsystem mechanization options: Several options were considered in choosing the type of modulation for the BVS-to-Earth telemetry link. In addition to the baseline system described above (a single biphase modulated square wave subcarrier phase modulating a carrier) two alternative approaches were considered -- use of block coding or use of noncoherent M'ary frequency shift key.

The use of block coding such as a (32, 6) comma free, biorthogonal code can nearly double the data rate for the same transmitter power. Although this is a perfectly feasible approach it is being reserved for a growth capability, since use of coding requires the design and manufacture of terminal equipment as well as the planning for and installation of the equipment at the NASA deep space net stations. The cost may not be warranted for this application.

The M'ary FSK approach for the baseline transmitter power and other link parameters is not competitive since the crossover point on transmitter power occurs near a bit rate of 5 bps (the MFSK begins to excel at the lower rates). A further disadvantage of incoherent MFSK is the complications in obtaining two-way doppler and ranging in the standard manner used by NASA in their Deep Space Net operations. A comparison of 5-bit incoherent MFSK, PCM/PSK/PM, and coded PCM/PSK/PM is given in figure 61. The data were taken from reference 8 and modified for the conditions for this mission.

Another option is to eliminate the frequency search and acquisition mode for the BVS receiver; however, elimination of this feature would be expensive in electrical energy wasted by the BVS transmitter during acquisition operations by the Deep Space Net.

A comparison of acquisition methods follow. For the nonsearch BVS receiver case the Earth-based transmitter must be swept in frequency until the BVS receiver APC loop frequency pull-in range is encountered. Locking to the Earth-transmitted frequency causes the BVS transmitter to be activated. But how does the Deep Space Net operator know when the BVS receiver frequency is found? He must detect the BVS transmitted signal. This means that (assuming he detects a signal from the BVS) the signal he receives corresponds to frequency translation of the Earth-based transmitter frequency (modified by doppler), which was being used approximately 10 minutes before detection of the signal (the round trip delay time). One approach is to then estimate the frequency, return to that region and wait another 10 minutes for confirmation. Under these conditions it is not hard to predict a BVS transmitter on-time of 20 minutes or more before the Deep Space Net can acquire two-way carrier lock and begin demodulating data.

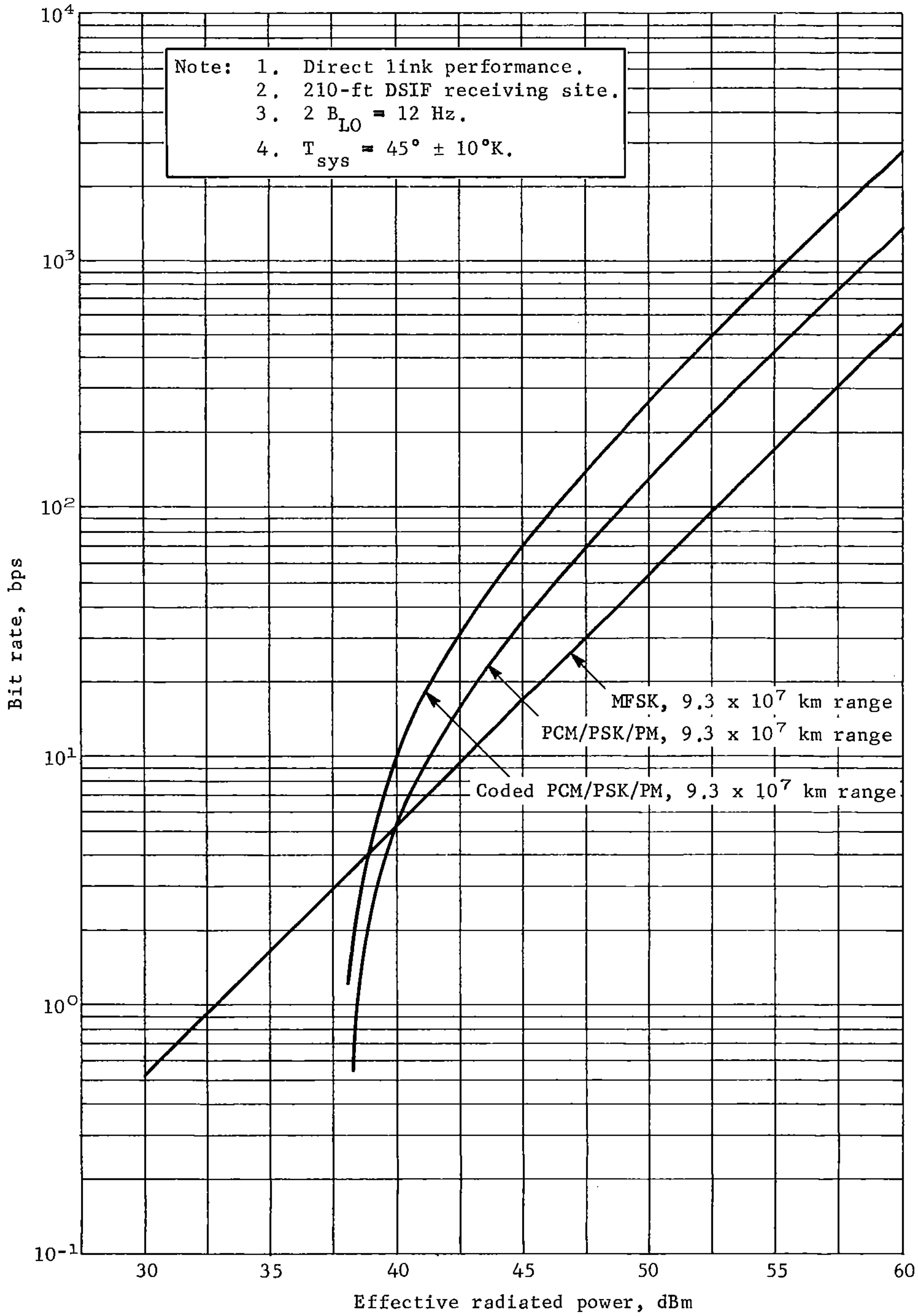


Figure 61.- Bit Rate versus Effective Radiated Power

On the other hand if the BVS receiver has a search mode, the DSN transmitter frequency can be set to within the known search range of the BVS receiver. Lock-on time for the BVS receiver is approximately 87 sec after the signal reaches the vicinity of Venus assuming a maximum sweep rate of 490 Hz/sec. Five minutes later the BVS signal reaches Earth and within a few seconds (possibly 30) two-way carrier lock is established.

The rapid lock-on to the BVS signal occurs since the translated frequency of the received signal should be known to within a few hundred doppler cycles.

Thus, for initial acquisition contact (with a BVS receiver search mode and two-way coherent operation) the BVS transmitter on-time consumed in carrier acquisition is only a few seconds. It therefore appears that there is little choice but to include a search mode in the BVS receiver to prevent electrical energy from being wasted during acquisition operations.

The use of a postdeploy relay link through the spacecraft to Earth was investigated for communications with the spacecraft for several hours as the objective. It was determined that the capsule entry point must be on the side opposite the preencounter flight path. This condition resulted in a need for a capsule attitude reorientation subsystem. Further, the available postdeploy communications time was only 4 to 5 hr at 10 bps, even with a 20 W transmitter and use of coherent PSK/PM/PM modulation. The postdeploy relay link was therefore abandoned in favor of the direct link to Earth.

Command subsystem.- The BVS command subsystem is operative only after deployment of the BVS has been completed. Commands are received directly from Earth via the S-band radio subsystem.

Requirements: BVS command subsystem requirements are shown in tables 47 and a command list is shown in table 48.

Description: The command subsystem is made up of a command detector and a command decoder with interfaces as shown in the block diagram, figure 62.

Since the proposed operation of a typical command detector is identical to that of the Mariner 1969 unit, the block diagram (fig. 63) and description are taken unchanged from reference 9.

TABLE 47.- BVS COMMAND SUBSYSTEM REQUIREMENTS

1. Accept a composite signal from the radio subsystem containing the command word information and synchronization information.
2. Acquire phase coherence with the pseudorandom synchronization signal and establish a phase reference signal and bit synchronization signal.
3. Demodulate the command word information and reconstruct the command word data bits.
4. Decode the command word data bits and provide discrete momentary switch closures to the user subsystem.
5. Provide a capability for 8 commands.
6. Be compatible with the NASA Deep Space Net command standards (Mariner system).

TABLE 48.- BVS COMMAND LIST

1. Release drop sonde (after fixed delay).
2. Maximum transmitter on-time, X minutes.
3. Maximum transmitter on-time, Y minutes.
4. Activate ranging channel.
5. Deactivate ranging channel.
6. Modify science sampling interval.
7. Revert to normal science sampling interval.
8. Spare.

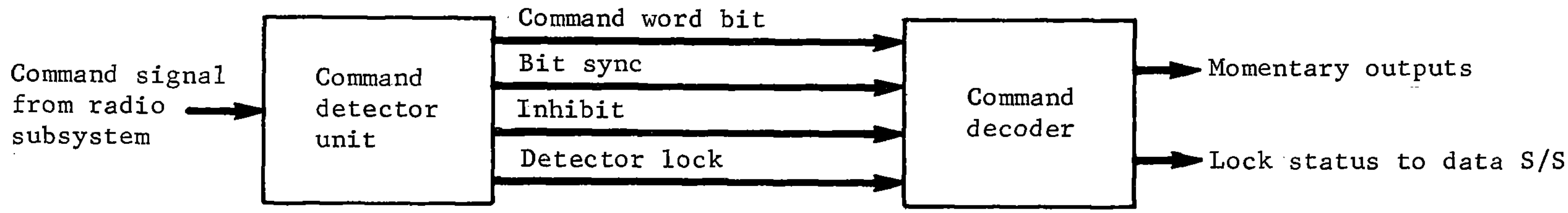
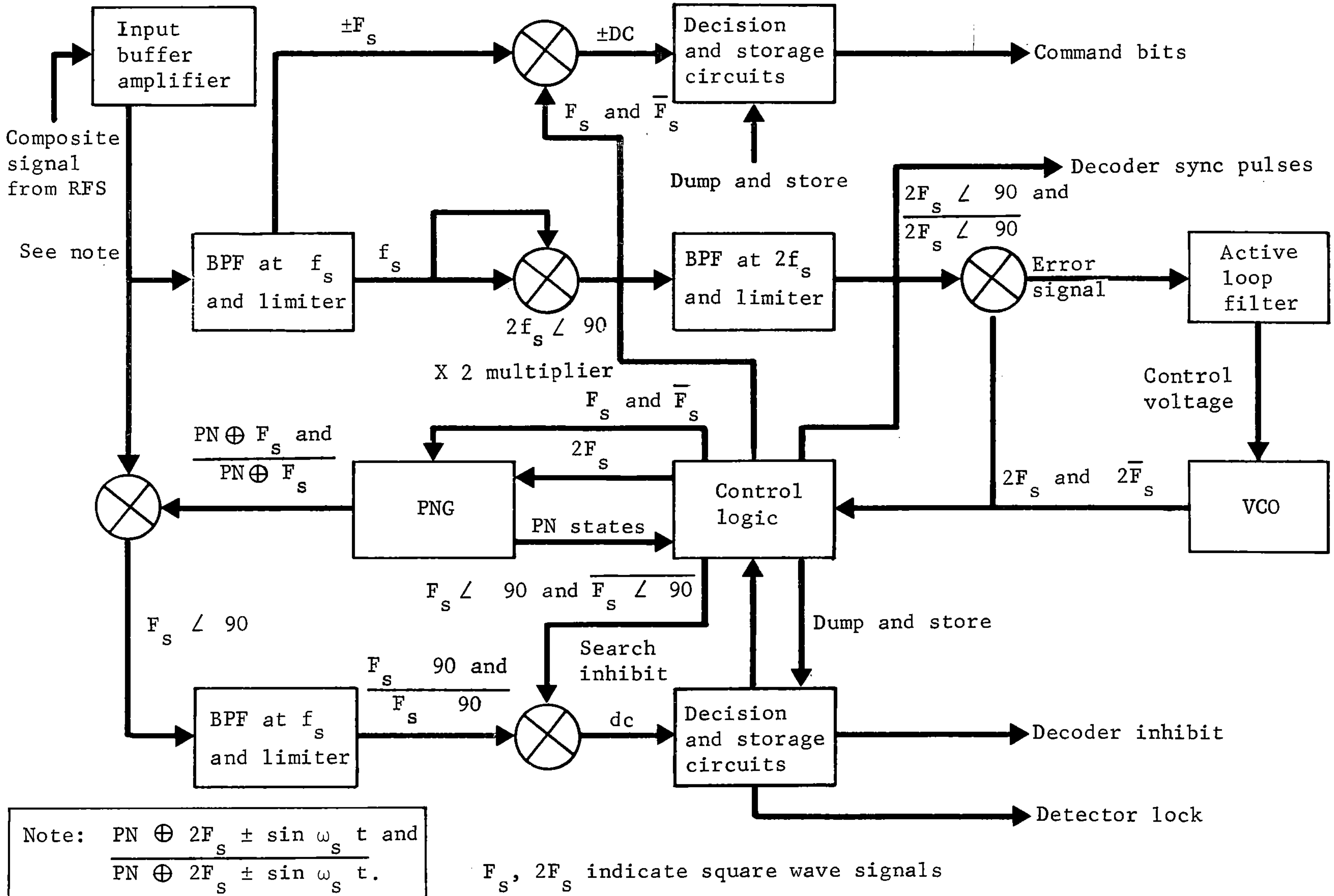


Figure 62.- Command Subsystem Functional Diagram



F_s, $2F_s$ indicate square wave signals
f_s, $2f_s$ indicate sinusoidal signals

Figure 63.- Command Detector, Functional Block Diagram

"3.3 Command Detector

"3.3.1 A functional block diagram of the command detector is presented in figure 63. The detector shall provide three separate and main functions as follows:

"a. Demodulate the biphase modulated sinusoidal subcarrier from the composite command signal by means of the detector command word channel.

"b. Establish a phase reference signal for use in demodulating the sinusoidal subcarrier and establish bit synchronization by means of the synchronization channel.

"c. Generate a detector lock signal which indicates whether or not the detector is in lock with the command synchronization signal, to be accomplished by the detector lock channel.

"3.3.2 Detector Command Word Channel. The detector word channel will select the command word information signal (biphase modulated sinusoidal subcarrier) from the composite command signal by means of a bandpass filter at the subcarrier frequency (figure 63). The subcarrier will then be phase coherently demodulated through multiplication of the subcarrier by a phase reference signal from the detector sync channel and through filtering of the multiplied signals by a matched RC filter. The output of the matched filter is then applied to a decision circuit and storage circuit which reconstructs the command word binary digit. The recovered command word bits are then directed to the command decoder for processing and decoding.

"3.3.3 Detector Sync Channel. The detector sync channel will separate the sync information from the input composite command signal by a process of filtering and multiplication. The sync signal is comprised of PN $\oplus 2f_s$, where PN is a pseudo-random binary sequence and $2f_s$ is the PN clock frequency. The sync channel circuitry will consist of a phaselock loop which will establish phase coherence with the incoming signal by locking a voltage controlled oscillator to the subcarrier frequency, and a PN generator and associated logic circuitry. This shall generate a pseudo-random binary sequence identical to that included in the ground transmitted command signal. When the detector-generated PN sequence and the ground-transmitted PN sequence are exactly coincident, established by a cross-correlation process, the detector bit sync will be coincident to the ground modulator bit sync and detection of the command data bits will be possible. The PN generator shall be clocked by the $2f_s$ frequency with the length of the random binary sequence equal to L, where $L = 2^n - 1$ (n is an integer). A bit sync pulse will be derived from the PN generator every complete cycle of the PN sequence, with the time between bit sync pulses (1 sec) being the duration of the command bits.

"3.3.4 Detector Lock Channel. The detector lock signal shall be derived from the sync signal portion $(PN \oplus 2f_s)$ of the composite command signal by a process of correlation and filtering. A voltage will be developed, after correlation and filtering, which is positive when the detector PN sequence is not coincident with the ground PN sequence and negative when they are coincident. This voltage level will then be applied to a matched filter circuit and decision element which interpret the polarity and indicate the lock status of the detector."

A functional block diagram of the command decoder is shown in figure 64. The main functions of the decoder are command word processing, decoding, and command word output execution.

Only direct commands are processed. The command word consists of 26 bits to conform to the Mariner standard. The first three bits are fixed and provide command decoder start. The next 6 bits are fixed and indicate a direct command (if properly coded) with an even number of "one" bits. The following 10 bits are all zero and the last 7 bits are the user address, which must contain an even number of "one" bits.

When the proper word start code is recognized the decoder will examine the next 6 bits to see if a direct command has been sent. If these are coded properly it will examine the last 7 bits to determine the correct address and initiate a momentary closure of the proper output switch.

When the detector lock signal indicates a detector out-of-lock condition, the decoder is inhibited until 26 consecutive zeros have been received. This prevents operation of the decoder while the command detector's subcarrier and sync channels are not synchronized.

Table 49 estimates size, weight, and power requirements for the command systems. Development status is also shown.

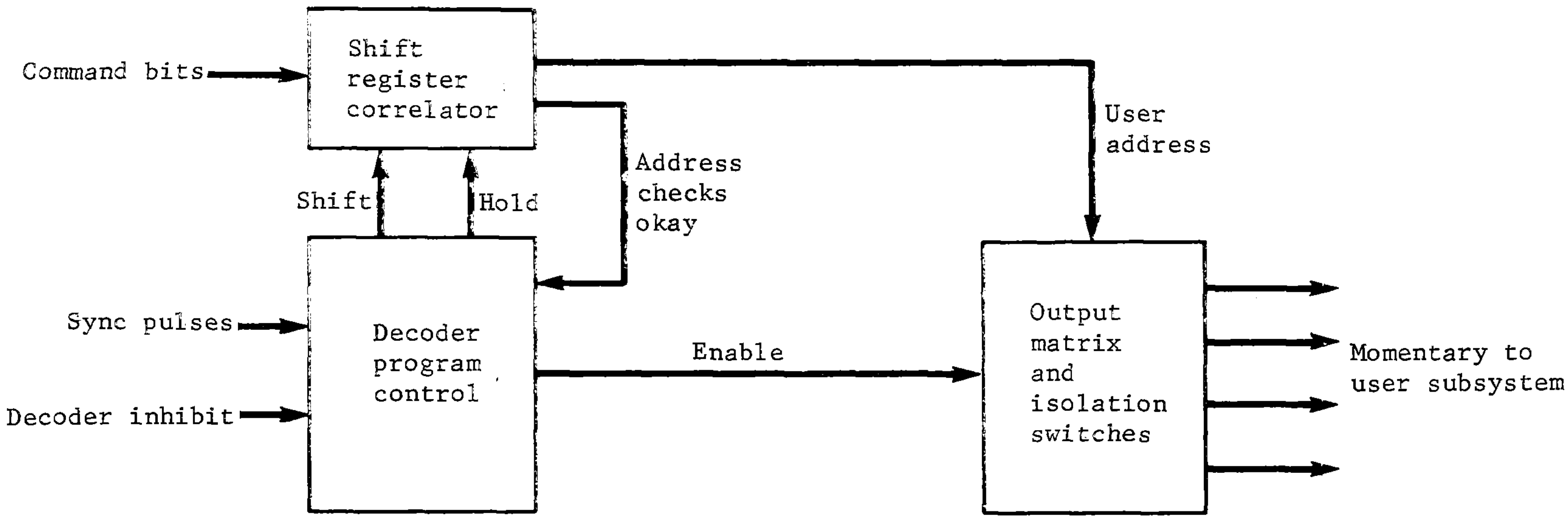


Figure 64.- Command Decoder, Functional Block Diagram

TABLE 49.- BVS COMMAND SUBSYSTEM SIZE, WEIGHT, AND POWER

Component	Weight, lb	Power, W	Size, cu in.	Development status
Command detector	4	1.5	40	Same circuit design as Mariner 1969
Command decoder	3	6	60	Uses Mariner 1969 word format but only direct command portion of circuitry.

Performance: The command detector threshold is defined as the value of command signal power-to-noise ratio that causes a command word bit error rate of less than or equal to 1×10^{-5} errors per bit. Table 50 is a design control table showing link parameters for conditions above threshold. Note that adequate margin is provided with use of the 85-ft DSIF transmitting antenna and the 100 kw command transmitter. (The BVS-to-Earth link reception requires use of the 210-ft diam receiving antenna.)

At least two bit errors must occur before or within a transmitted command word before the wrong command can be executed. The maximum probability of an incorrect command word being executed assuming threshold conditions (ref. 9) is:

$$P_{WE_{max}} = 1 \times 10^{-6}$$

Mechanization options: The baseline system described above conforms to the NASA DSN standard for command systems and if used will require no modification to the DSN. The command detector circuitry has been developed and used on the Mariner spacecraft and applicable portions of the decoder circuitry will soon be used on Mariner 1969.

No other approach has been considered due to the standardization of the system.

BVS data subsystem.- The BVS data subsystem functional requirements include controlling power to the science instrumentation, providing the sequencing control associated with the collection, processing, and transmission of data, as well as performing the commutation of data, the conversion of data, generation of time, generation of data formats, and the storage and playback of data.

TABLE 50.- COMMAND LINK DESIGN CONTROL TABLE

Parameter	Value	Tolerance +	Tolerance -
Total transmitter power, 100 kw	+80.0 dBm		
Transmit circuit loss	-0.4 dB	0.1	0.1
Transmit antenna gain (85 ft)	+51.5 dB	0.8	0.8
Transmit antenna pointing loss	-0.3 dB	0.3	0.0
Space loss, F = 2115 MHz, ρ_c = 93 x 10^6 km	-257.6 dB	----	----
Polarization loss	-0.6 dB	0.6	0.0
Receiving antenna gain	+5.0 dB	0.0	0.5
Receiving antenna pointing loss	-3.5 dB	3.5	1.5
Receiving circuit loss	-1.5 dB	0.2	0.2
Net circuit loss	-207.4 dB	5.5	3.1
Total received power	-127.4 dBm	5.5	3.1
System noise spectral density, T = 4070°K	-162.5 dB	4.0	0.0
Carrier modulation loss	-3.2 dB	0.5	0.6
Received carrier power	-130.6 dBm	6.0	3.7
Carrier APC noise BW, 2 B_{LO} = 70 ~	+18.5 dB	0.5	0.4
2 B_{LO} = 20 ~	+13.0 dB		
<u>Carrier performance - command</u>			
Threshold SNR in $2B_{LO}$	+8.0 dB	1.0	1.0
Threshold carrier power (for 2 B_{LO} = 70 Hz)	-136.0 dBm	5.5	1.4
Performance margin for 2 B_{LO} = 70 Hz	+5.4 dB	11.5	5.1
Performance margin for 2 B_{LO} = 20 Hz	+10.9 dB	11.5	5.1
<u>Data Channel</u>			
Modulation loss	-8.5 dB	0.5	0.5
Received subcarrier power	-135.9 dBm	6.0	3.6
PLL noise bandwidth (BW = 1 Hz)	0.0 dB	----	----
Threshold SNR (P_E = 10^{-5})	+15.7 dB	1.0	1.0
Threshold subcarrier power	-146.8 dBm	5.0	1.0
Net margin	+10.9 dB	11.0	5.6
<u>Sync channel</u>			
Modulation loss	-5.5 dB	0.7	0.7
Received subcarrier power	-132.9 dBm	6.2	3.8
Threshold SNR	+15.7 dB	1.0	1.0
Noise bandwidth 2 B_{LO}	+3.0 dB	1.0	0.8
Threshold channel power	-143.8 dBm	6.0	1.8
Net margin	+10.9 dB	12.2	5.6

During the period from 5 minutes before separation of the capsule from the spacecraft to separation of the BVS from the aeroshell, the BVS data system must accept aeroshell sensor data inputs from an aeroshell multiplexer as well as control its operation. Before this period (during interplanetary cruise) the aeroshell data multiplexer is under the control of the spacecraft subsystems.

Following station deployment the subsystem must generate and biphase modulate the data subcarrier for the S-band direct link to Earth.

Table 51 lists the engineering and science data requirements for the BVS as well as the required sampling rate. The measurements are grouped into formats depending on the time period they are required.

Table 52 lists the data modes and the periods of time they are used. Requirements for data subsystem programer inputs and outputs are shown in table 53.

Subsystem description: The data subsystem consists of a data multiplexer/encoder, memory, transducer power supply, and a data subsystem sequencer, all located in the BVS. A remote multiplexer and a transducer power supply are located in the aeroshell. A block diagram of the subsystem is shown in figure 65. All science and engineering data are processed through this data subsystem.

The data subsystem sequencer (DSS), as for the orbital mission, provides all the timing and power control required on the BVS for the sequencing of experiments, data subsystem equipment, command, and radio equipment to collect, process, store, and transmit data. It accepts control signals from the aeroshell sequencer and the BVS sequencer before full BVS deployment. Following this event the data subsystem sequencer provides all station sequences until the end of the mission. Reset for science sampling schedules occurs when the DSN makes its scheduled contact with the station.

The data multiplexer/encoder accepts 0 to 40 MV and 0 to 5 Vdc analog signals, bilevel and digital signals, provides a digital signal interface with the memory, and provides control of the aeroshell multiplexer. A hybrid level signal input for analog signals is selected to minimize the signal conditioning and the transducer power requirements. The format is generated by this unit for those data that are sampled at regular intervals from various instruments.

TABLE 51.- BVS/ENTRY VEHICLE DATA REQUIREMENTS

Measurement	Bits/sample	Format and sample rate (sps)			
		A	B	C (stored)	C (real time)
Temp, aeroshell 1	8	½	½		
Temp, aeroshell 2					
Temp, aeroshell 3					
Temp, backshell 1					
Temp, backshell 2	8				
Ablator thickness 1	3				
Ablator thickness 2					
Ablator thickness 3					
Ablator thickness 4					
Ablator thickness 5	3				
Ablator thickness 6	1	½	½		
Temp, gondola 1	8	¼	¼	1/hr	1/32
Temp, gondola 2				1/hr	1/32
Temp, fuel tank					
Pressure fuel tank					
Pressure gas tank					
Temp, inflation tank					
Temp, Ag-Zn battery BVS				1/hr	1/32
Voltage, Ag-Zn battery BVS				1/hr	1/32
Temp, probe battery					
Voltage, probe battery					
Voltage, sonde battery 1				1/hr	1/32
Voltage, sonde battery 2				1/hr	1/32
Temp, aeroshell battery					
Voltage, aeroshell battery					
Voltage, transducer supply aeroshell					
Voltage, transducer supply BVS				1/hr	1/32
Voltage, BVS main bus				1/hr	1/32
Current, BVS main bus				1/hr	1/32
Power, BVS transmitter (relay link)					
Temp, BVS transmitter (relay link)		¼	¼		
Pressure (thrust chamber)		1	1		
Accelerometer longitudinal (low)		1	1		
Accelerometer longitudinal (high)		--	20		
Accelerometer lateral		--	20		
Accelerometer vertical	8	--	20		
Aeroshell discretes, 8 each	8 (1 each)	2	2		
Aeroshell discretes, 8 each	8 (1 each)	¼	¼		
Aeroshell discretes, 8 each	8 (1 each)	¼	¼		
Aeroshell discretes, 8 each	8 (1 each)	¼	¼		
BVS discretes, 8 each	8 (1 each)	2	2	1/hr	1/32
UV photometer aeroshell 1	8	2	2		
UV photometer aeroshell 2		2	2		
Pressure inflation orifice BVS		¼	¼		
Pressure balloon BVS				1/hr	1/32
Temperature balloon BVS					
Strain balloon film BVS 1					
Strain balloon film BVS 2					
Reference temperature					
Pressure, atmospheric (low)					
Pressure, atmospheric (high)					
Temp, atmosphere 1					
Temp, atmosphere 2					
Solar aspect sensor 1					
Solar aspect sensor 2					
Solar aspect sensor 3					
Water vapor detector					
Visual photometer BVS 1					
Visual photometer BVS 2					
Visual photometer BVS 3	8	¼	¼	1/hr	1/32

TABLE 51.- BVS/ENTRY VEHICLE DATA REQUIREMENTS - Concluded

Measurement	Bits/sample	Format and sample rate (sps)			
		A	B	C (stored)	C (real time)
Radar altimeter 1	8	¼	¼	1/hr	1/32
Radar altimeter 2	4				
Radar altimeter 3	4				
Radar altimeter 4	4				
Radar altimeter 5	4				
Frame sync	21				
Frame identity	3	¼	¼	1/hr	1/32
Sub frame sync	8	1	1	4/hr	1/8
Time	16	1	1	4/hr	1/8
Spares (7 words)	56 (8 each)	¼	¼	1/hr	1/32
Spares (6 words)	48 (8 each)			2/hr	1/16
Spares (engineering status) (3 words)	24 (8 each)			8/hr	1/4
Receiver AGC fine	8			1/hr	1/32
Receiver AGC coarse					
Crystal oscillator temp					
Exciter voltage					
Exciter output drive					
TWT base temperature					
TWT helix voltage					
TWT helix current					
VCO current					
VCO temperature					
Command decoder current					
Detector lock (included in discretes)				1/hr	1/32
Local oscillator drive				2/hr	1/16
Static phase error				2/hr	1/16
TWT power out	8			2/hr	1/16
Mini bio lab	48			1/hr	1/32

Engineering and Science Data Requirements

Measurement	Rate	Format
Mass spectrometer Emission current Filament current Repeller potential Accelerating potential (min.) Accelerating potential (max.) Electrostatic sector reference potential SEM reference potential Ion pump current	Formatted for one frame (1200 bits) of data once each 8 hr. (20-sec sample time)	G
Gas chromatograph	Formatted into one frame (1200 bits) of data once each 8 hr. (20-minute sample time)	H
Accelerometer, longitudinal, low range Accelerometer, X axis, low range Accelerometer, Y axis, low range	24-1/3 sec of data formatted with sync and time (1200 bit frame) once each 8 hr if pre-set accelerometer threshold is exceeded.	I
Status monitor (aeroshell)	Selected BVS/entry vehicle data are sampled and included in spacecraft T/M during cruise.	J
Sonde data Pressure, atmospheric Temperature, atmospheric Water vapor Voltage, battery Temp, electronics Frame count	Formatted in sonde (240 bit frame every 4 minutes until impact).	K

TABLE 52.- TYPICAL BVS DATA MODES

Mode	Period used and description
A	Real-time data are collected and transmitted during capsule/spacecraft separation through completion of ΔV burn and during entry through deployment of station except for 5 minutes when the BVS is suspended from its parachute immediately after separation from the aeroshell. Consists of repetitive frames of aeroshell and BVS engineering plus science data. See Format A for content as shown in table 51.
B	Transmitted for 5 minutes beginning at 1 minute after separation of the BVS from the aeroshell. Consists of repetitive frames of engineering data stored during the entry blackout and high g. Data content conforms to Format B as shown in table 51.
C	Collected and stored on an hourly basis following deployment of station. Consists of engineering and science data as shown in table 51 for Format C. Transmitted to Earth on direct link after first collection cycle and approximately every 8 hr thereafter during mode E.
D	Collected and stored immediately after station deployment. Collected on the 6th hourly cycle and each 8 hr thereafter. Consists of two full frames of Format C plus a 1200-bit frame of each of mass spectrometer Format G, gas chromatograph Format H, and accelerometer Format I. (Gas chromatograph frame is not collected in time to be transmitted immediately after deployment of station but is sampled during this period and stored following the first Earth link data transmission.) Data are transmitted to Earth during mode E.
E	Mode E is a data transmission mode. It occurs at the time of contact with the Earth station. When the BVS S-band transmitter is turned on by receipt of an Earth signal real-time frames of Format C are transmitted for 4 minutes. This is followed by repeated playback of all stored data that were collected during modes C, D, and F, and ends with several real-time frames of Format C to confirm receipt of any commands from Earth. The total transmission period is 20 minutes.
F	Mode F is a sonde data collection and storage cycle. It consists of 30 minutes of receiving and storing data transmitted from the sonde to the BVS station main storage. The data are transmitted directly to Earth during mode E operations. The data format is identified as Format K.

TABLE 53.- DATA SUBSYSTEM PROGRAMER (SEQUENCER) INPUT/OUTPUT DISCRETES

Input discretes (for control)	From
1. Start mode A operation	Aeroshell sequencer
2. Start mode B storage write	Aeroshell sequencer
3. End mode B storage write	BVS sequencer
4. End mode A operation	Aeroshell sequencer or BVS sequencer
5. Begin mode B storage read	BVS sequencer
6. End mode B storage read	BVS sequencer
7. Begin flotation operating modes	BVS sequencer
8. Initiate release of drop sonde sequence	Command decoder
9. Maximum S-band transmitter on-time 20 minutes	Command decoder
10. Maximum S-band transmitter on-time Y minutes	Command decoder
11. Begin mode E operation	Command decoder
Output discretes	**To**
1. Transmitter on/off	BVS power control
2. Data multiplexer/encoder, transducer power supply, data memory, atmospheric science instruments on/off	BVS power control
3. Bio lab on/off	BVS power control
4. Mass spectrometer on/off	BVS power control
5. Gas chromatograph	BVS power control
6. Main receiver on/off	BVS power control
7. Command detector and decoder on/off	BVS power control
8. Sonde data unit on/off	BVS power control
9. Arm sonde	BVS power control
10. Release sonde	BVS power control
11. Safe sonde	BVS power control
12. Accelerometer on/off	BVS power control
13. Radar on/off	BVS power control
14. thru 20. Formats (6) start/stop	Data multiplexer/encoder
21. Atmospheric sample collection	Science
22. Bio sample collection	Science
23. thru 28. Science discretes (mode control)	Science
29. thru 34. Data handling discretes (mode control)	Data multiplexer/encoder

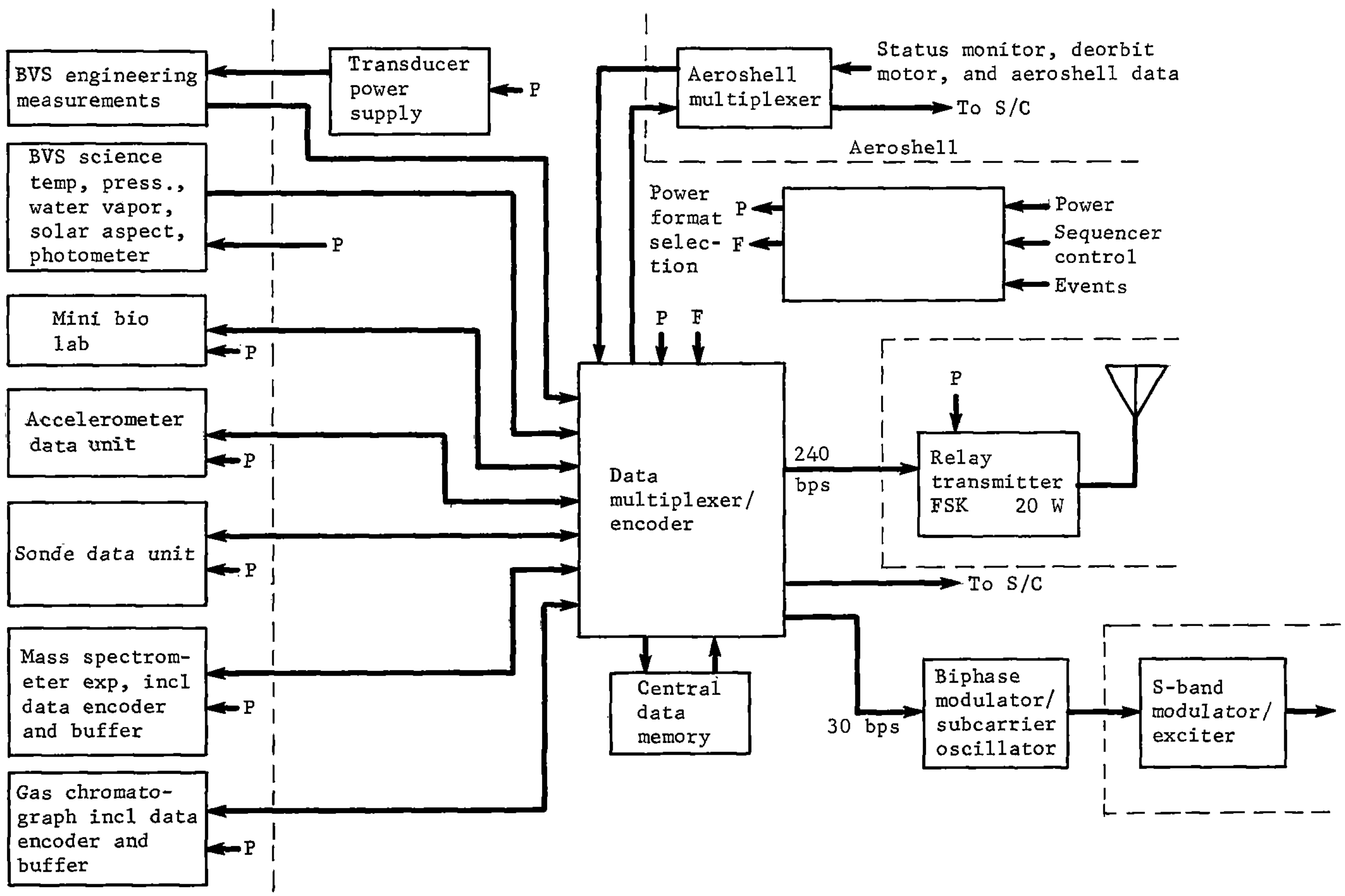


Figure 65.- BVS Data Handling Subsystem

Experiments with unique data handling requirements, such as the gas chromatograph, mass spectrometer, sonde, and the accelerometer for monitoring wind gusts have their own associated data processing electronics, as is the case for the orbital mission, and provide a digital data interface with the data multiplexer/encoder. Formatting and buffer storage of an experiment sample is accomplished in each experiment data unit. Clocking and control is provided by the data subsystem sequencer. Data from the various experiment buffers is sequenced into the data memory under control of the DSS.

The memory is sized by the requirements to store data during radio blackout and during peak g levels at entry. This same memory is used to store data between direct-to-Earth communication contacts after deployment of the station. Memory capacity is the same as for the orbital mission.

The transducer power supply provides 5 Vdc regulated power to those engineering sensors requiring regulated power.

Size, weight, and power required for each unit of the data subsystem are shown in table 54.

TABLE 54.- DATA SUBSYSTEM SIZE, WEIGHT, AND POWER

Component	Size, cu in.	Weight, lb	Power, W
BVS			
Data subsystem sequencer	280	4.0	4.0
Data multiplexer/encoder	420	10.0	8.5
Memory	200	4.0	1.0
Transducer power supply	20	1.0	1.0
Total	920	19.0	14.5
Aeroshell			
Multiplexer/encoder	280	3.0	4.0
Transducer power supply	20	0.5	1.0
Engineering sensors	80	2.0	----
Total	380	5.5	5.0

Subsystem operation: The data subsystem operation is essentially the same as for the orbital mission from launch to immediately preceding entry after which the main difference is in data mode and postdeploy operations schedules.

At entry minus 20 minutes the aeroshell sequencer sends a discrete to the DSS that activates the data subsystem and 390 MHz transmitter. Data Format A (as previously defined) is transmitted to the spacecraft until immediately after separation of the BVS from the aeroshell. For the next few minutes, while the BVS is decelerating, the accelerometer data and other engineering data that were recorded following entry are transmitted to the spacecraft. The data transmission reverts to real time before station deployment occurs. At completion of deployment when the inflation tankage, the BVS sequencer, and the 390 MHz transmitter and antenna are released and fall to the surface, all BVS data functions are under control of the BVS data subsystem sequencer and the command subsystem.

The data collection and transmission routines following station deployment are based on an hourly sampling of the more routine science instruments plus engineering, a once-per-8-hr sampling of all science instrumentation, and transmission of real-time plus stored data on a once-per-8-hr cycle assuming the Earth-based stations turn the BVS transmitter "on" on an 8-hr cycling basis.

The once-per-8-hr sampling is scheduled to be taken at the sixth hourly period following deployment, and then is taken on an 8-hr recurring cycle.

The DSS unit schedules turn on of the S-band receiver for 5 minutes at the beginning of each hourly collection period. If the receiver finds a carrier present, it will turn on the BVS transmitter and remain activated for 20 minutes. This allows the DSN station to make contact with the station to initiate a "release drop sonde" command. The number of station contacts allowable is limited by the electrical energy that can be stored in the battery systems; therefore, making contact more than once per 8-hr can jeopardize maximum mission lifetime. By issuing a "release drop sonde" command on only the regularly scheduled contacts one can conserve the station batteries. Sonde release, when commanded, is delayed until the end of BVS data transmission to reduce peak power load in the system and to free the data system from its data transmission routine.

Mechanization options: The major tradeoff considered in the data subsystem is in the use of a decentralized versus a centralized data subsystem. A decentralized approach was taken for this mission for the same reasons as it was taken for the orbital mission.

Aeroshell mounted data subsystem.- The aeroshell data subsystem is essentially the same as for the orbital mission, consisting of a data multiplexer/encoder and a transducer power supply. Size, weight, and power of the data equipment mounted in the aeroshell are shown along with the BVS data system components in the previous section.

Telecomunications for Mercury/Venus Mission

BVS radio and command subsystems.- Requirements for the BVS/ entry vehicle and command subsystems are the same as for the flyby mission except that the S-band direct link to Earth is used for preentry, entry, and postentry communications because no relay link to the spacecraft is used. The data rate is 120 bps for all phases of operation. This represents a reduction in the data rate from 240 to 120 bps for the entry and an increase in data rate from 30 to 120 bps for the postentry operations. An increased data rate can be used because of the shorter range to Earth than for the flyby mission. The command list is identical to the flyby mission list.

The S-band radio subsystem components are identical to those of the Venus flyby mission except the BVS S-band receiver carrier tracking loop $2B_{LO}$ bandwidth can be increased from 70 Hz to 250 Hz to provide a higher frequency search sweep rate for decreased acquisition time on Earth-to-BVS link. The 370 MHz relay transmitter and antenna used for the Venus flyby relay link to the spacecraft are deleted.

Radio and command subsystem performance: During separation of the capsule from the spacecraft and during the deflection burn phase of the mission, telemetry data will be recorded in the BVS data storage. The S-band radio system will then be activated and real-time plus stored data will be transmitted directly to Earth from the capsule for 20 minutes. The transmitter will be turned off during the capsule coast phase then reactivated for the entry and deployment phases.

Following deployment of the station, the science sampling schedule and data transmission schedule will be identical to that of the Venus flyby mission.

A design control table for the S-band telemetry link and the S-band command link (both direct links with Earth) are given in tables 55 and 56, respectively. Note that the margins are adequate to provide a telemetry bit rate of 120 bps while using a 48 Hz carrier. tracking loop bandwidth for the DSN receiver and to provide commands while using a BVS carrier tracking loop bandwidth of 250 Hz.

A range to Earth of 46.75×10^6 km was chosen as maximum. This corresponds to a range at 5 days after entry. Since range increases as a function of time for the period of interest, the range entry plus 5 days represents a worst case.

TABLE 55.- 1973 TYPE I DIRECT LINK PERFORMANCE TELECOMMUNICATION DESIGN CONTROL TABLE

Parameter	Value	Tolerance (dB) +	Tolerance (dB) -	Notes
Total transmitter power	+43.0 dBm	1.0	0.0	20 W minimum
Transmitting circuit loss	-1.0 dB	0.5	0.5	
Transmitting antenna gain	+5.0 dB	0.5	0.5	
Transmitting Antenna Pointing Loss	-3.5 dB	3.5	0.0	$\pm 70°$
Space loss	-252.3 dB	---	---	
F=2295 MHz, ρ_c=46.75 x 10^6 km (0.32 A.U.) polarization loss = 6 dB, DSIF = 0.8 dB	-0.6 dB	---	---	Maximum
Receiving antenna gain	+61.0 dB	1.0	1.0	EPD-283
Receiving antenna pointing loss	-0.3 dB	0.3	0.1	+0.02°
Receiving circuit loss	-0.2 dB	0.1	0.0	estimate
Net circuit loss	-191.9 db	5.9	2.1	
Receiver noise spectral density T system = 45 $\pm$ 10°K	-182.1 dBm/Hz	1.1	0.9	EPD-283
Carrier modulation loss	-6.0 dB	0.4	0.4	θ = 1.05 rad
Received carrier power	-154.9 dBm	7.3	0.5	
Carrier APC noise BW 2 B_{LO} = 48 Hz	+16.8 dB	0.5	0.0	MC-4-310A
<u>Carrier performance - telemetry</u>				
Threshold SNR in 2 B_{LO}	+6.0 dB	0.5	1.0	MC-4-310A
Threshold carrier power	-159.3 dB	2.1	1.9	
Performance Margin	+4.4 dB	9.4	4.4	
<u>Data Channel</u>				
Modulation loss	-1.2 dB	0.4	0.4	θ = 1.05 rad
Received data subcarrier power	-150.1 dBm	7.3	2.5	
Bit rate (1/T = 120 bps)	+20.8 dB	---	---	
Required ST/N/B	+6.8 dB	1.0	1.0	$P_e^b = 5 \times 10^{-3}$
Threshold subcarrier power	-154.5 dBm	2.1	1.9	
Performance margin	+4.4 dB	9.4	4.4	

TABLE 56.- COMMAND LINK DESIGN CONTROL TABLE

Parameter	Value	Tolerance, dB +	Tolerance, dB -
Total transmitter power, 100 kw	+80.0 dBm		
Transmit circuit loss	-0.4 dB	0.1	0.1
Transmit antenna gain (85 ft)	+51.5 dB	0.8	0.8
Transmit antenna pointing loss	-0.3 dB	0.3	0.0
Space loss			
$F = 2115$ MHz, $\rho_c = 46.75 \times 10^6$	-251.6 dB	---	---
Polarization loss	-0.6 dB	0.6	0.0
Receiving antenna gain	+5.0 dB	0.0	0.5
Receiving antenna pointing loss	-3.5 dB	3.5	1.5
Receiving circuit loss	-1.5 dB	0.2	0.2
Net circuit loss	-201.4 dB	5.5	3.1
Total received power	-121.4 dB	5.5	3.1
System noise spectral density			
T = 4070°K	-162.5 dB	4.0	0.0
Carrier modulation loss	-3.2 dB	0.5	0.6
Received carrier power	-124.6 dBm	6.0	3.7
Carrier APC noise BW for 2 B_{LO} = 70 Hz	+18.5 dB	0.5	0.4
for 2 B_{LO} = 250 Hz	+24.0 dB	0.5	0.4
<u>Carrier performance - command</u>			
Threshold SNR in 2 B_{LO}	+8.0 dB	1.0	1.0
Threshold carrier power for 2 B_{LO} = 250 Hz	-130.5 dBm	5.5	1.4
Performance margin for 2 B_{LO} = 250 Hz	+5.9 dB	11.5	5.1
Performance margin for 2 B_{LO} = 70 Hz	+11.4 dB	11.5	5.1
Performance margin for 2 B_{LO} = 20 Hz	+16.9 dB	11.5	5.1
<u>Data Channel</u>			
Modulation loss	-8.5 dB	0.5	0.5
Received subcarrier power	-135.9 dBm	6.0	3.6
PLL noise bandwidth (BW = 1 Hz)	0.0 dB	---	---
Threshold SNR ($P_E = 10^{-5}$)	15.7 dB	1.0	1.0
Threshold subcarrier power	-146.8 dBm	5.0	1.0
Net margin	+16.9 dB	11.0	5.6
<u>Sync Channel</u>			
Modulation loss	-5.5 dB	0.7	0.7
Received subcarrier	-132.9 dBm	6.2	3.8
Threshold SNR	+15.7 dB	1.0	1.0
Noise bandwidth = 2 Hz	+3.0 dB	1.0	0.8
Threshold channel power	-143.8 dBm	6.0	1.8
Net margin	+16.9 dB	12.2	5.6

Antenna pointing loss is shown as 3.5 dB (the same as for the orbital mission). It represents an antenna aspect angle to Earth of 70° (20° above horizontal). This is the maximum desired limit. The initial BVS antenna aspect angle to Earth before entry is approximately 20° and during near vertical descent is approximately 50°. Predicted drift of the station is such that the aspect angle to Earth should improve initially as the station drifts closer to the subearth point.

Postdeploy acquisition and command procedures during floatation operations are identical to those described for the Venus flyby mission; however, the entry-through-deploy phase procedures are different since two-way S-band communications are also used during this period.

During entry the capsule experiences a deceleration of several hundred times earth gravity. The concern during this period is reacquiring the signal after loss of lock due to blackout and high doppler rates. Radio blackout is expected to occur for approximately 14 to 18 sec, ending before the maximum g force occurs. This means that both the up-link and down-link will be inoperative during this period.

The maximum tracking rate of the BVS receiver carrier loop will be exceeded until the deceleration has dropped below about 182 g, and assuming a BVS receiver frequency search rate of 6 Hz/sec, the g forces must be reduced to about 90 g before the BVS receiver can be expected to reacquire the signal from Earth. (Doppler rate plus search rate must not exceed the maximum allowable tracking rate, which is estimated at 0.17 times the $2B_{LO}$ carrier loop bandwidth squared.)

For the down-link it can be shown that a conservative estimate of the maximum tracking rate for the Earth-based receiver using a 4 Hz $2B_{LO}$ carrier loop bandwidth near threshold S/N ratios is about 400 Hz/sec (ref. 10). This rate corresponds to an effective g force of approximately 2.6 g acting on the entry capsule. Therefore it can be assumed that as the g forces are diminishing up-link lock will have reoccurred long before the two-way doppler rate will have dropped low enough to be tracked by the Earth-based receiver. However, since the frequency uncertainty is only the doppler offset, the Earth-based receiver (following loss of lock at entry) can be tuned to a frequency that will occur as the capsule velocity approaches that expected to occur near 1 g. The receiver should lock to the signal as the signal frequency sweeps through the receiver lock on range. If this fails then a frequency search must be made to acquire the signal while the BVS is hanging from the deploy parachute.

Broadband predetection recording at the DSN would be used so that the data may be recovered even though the real-time acquisition attempts fail.

Data recorded in the BVS during entry can be transmitted while the BVS is hanging on its deploy parachute. This prevents loss of data because of blackout and reacquisition periods in which real-time transmissions are interrupted.

Mechanization options: An optional approach to be considered during preentry-through-deployment phase is to use a one-way transmission mode to Earth instead of two-way coherent mode. For this mode the BVS transmitter frequency is determined by a self-contained crystal controlled oscillator instead of by the received command carrier. In this case both initial acquisition and reacquisition of the signal by the Earth station will be more difficult because of the BVS transmitter frequency uncertainty caused by temperature changes, aging, and shock. It could result in excessive loss of real-time entry data during the longer reacquisition period. These data may be recovered, however, with about 1 dB degradation (ref. 6) by processing the predetection recording made at the DSN Station.

Another option or alternative approach is to consider block coding on the BVS-to-Earth telemetry link as discussed for the orbital mission.

Aeroshell data subsystem.- The aeroshell data subsystem consists of a data multiplexer/encoder and transducer power supply, which are identical to those for the Venus flyby mission.

BVS data subsystem.- The BVS data subsystem requirements and configuration for the Venus/Mercury flyby mission are almost identical to the BVS Venus flyby mission. The exceptions are telemetry data rate plus minor changes in the telemetry channel assignments and power programing. The data modes are identical.

A fixed telemetry data rate of 120 bps is used for all data modes as opposed to the 240 bps for the preentry-to-deploy phases, and 30 bps for the poststation deploy phases of the Venus flyby configuration. The reason for this is use of a direct link to Earth for both the BVS/entry vehicle and for the BVS after separation from the capsule.

This change in data rate requires a reduction of telemetry channel sampling rates by one-half for the predeploy phase with no change in the multiplexer/encoder. Any critical channels can be cross strapped to spares and can be made available by reassigning lower priority channels to provide the required sampling rate. The postdeploy data rate increase is possible because of the lower Venus-to-Earth communications range for the Venus/Mercury flyby mission.

Data are recorded at 60 bps during capsule separation from the spacecraft and the recording continues through the deorbit engine burn phase of the mission. When the capsule is reoriented for entry and spun up immediately after the deorbit maneuver the data are transmitted directly to Earth. Entry accelerometer data are stored in memory at the same rate as for the Venus flyby, and thus are not affected by the change.

Minor power programing changes are encountered in use of the S-band system for entry in place of the relay link transmitter, but the effect of these changes are insignificant on the design of the subsystem.

BVS POWER AND PYROTECHNICS SUBSYSTEM

BVS Power Subsystem for the Orbiter Mission

Power is required to be furnished by a sterilized battery supplemented by solar cell panels. To hold the weight floated by the balloon to a minimum, the aeroshell-mounted battery is required to furnish power to the BVS to allow it to transmit data during separation from the spacecraft, ΔV burn, and conduct operations up to entry. The BVS battery is required to have sufficient capacity to support a mission of 50-hr duration. In addition, should light conditions be favorable, the mission time will be extended by using solar cells to replenish the battery charge. A monitor capability for detecting undervoltage is required. The undervoltage condition causes the equipment to be connected into a low power mode to give the solar cells additional time for recharging the batteries. Voltage levels at all subsystems are to be maintained between 24 and 36 V.

BVS power subsystem description.- As shown in figure 66 the BVS is equipped with a silver-zinc battery supplemented by solar cell panels. The solar cells are uniformly distributed on the perimeter of the gondola and are used to charge the battery during periods when peak loads are not present. The capacity of the solar cell panels is based on computed light values estimated to exist in the clouds. As long as sunlight is available, operation will be from the solar cell/battery combination. After the station drifts across the terminator, operation can still be carried out for an additional period of two orbits of the spacecraft.

The equipment, together with estimated weights and development status, is listed in table 57. The weights do not include supports or wiring and connections. An item of major interest is the development of sterilized silver-zinc batteries presently being carried out by JPL. Early in this program, it was concluded that the then existing batteries would not perform satisfactorily after heat sterilization because the materials used for plate separators degraded severely. Programs have been carried out on the syntheses of heat-sterilizable separator materials. Other studies in electrochemistry, cell fabrication, and sealing are almost completed. Currently underway are four hardware tasks:

1) A 5 A-h high impact cell;
2) 120 W-h and 600 W-h high impact, four-cycle-life batteries;
3) A 1200 W-h secondary battery;
4) A 2000 W-h four-cycle-life battery.

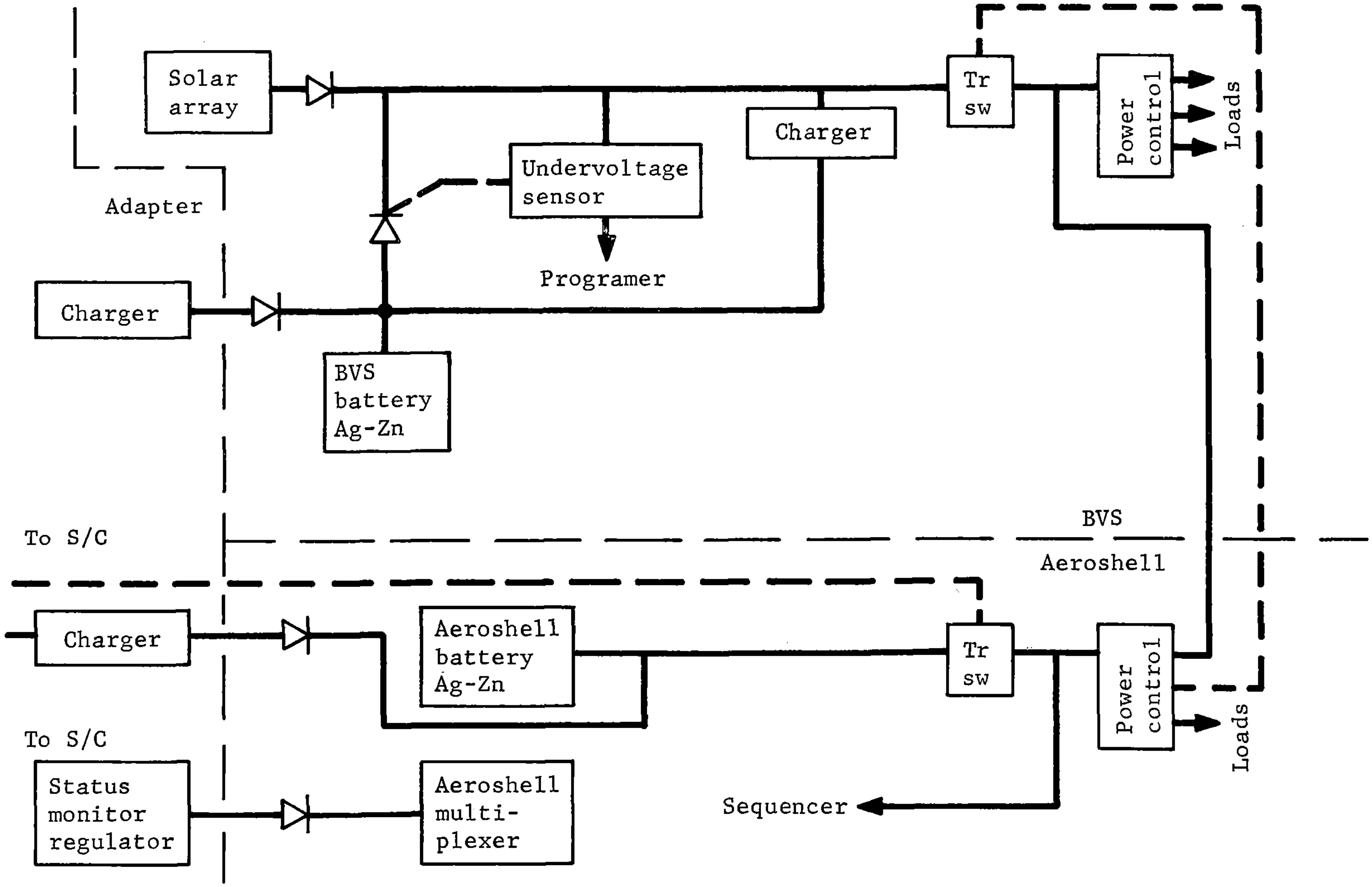


Figure 66.- Power Subsystem

TABLE 57.- BVS POWER SUBSYSTEM

Item	Weight, lb	Status
Ag-Zn battery	15.5	Under development
Solar panels	5.8	Current technology
Isolating diodes	1.0	Available
Charger	2.5	Current technology
Undervoltage sensor and SCR	.5	Current technology
Power transfer switch	1.0	Available
Power control module	3.0	Current technology
Auxiliary equipment	1.0	Current technology
Total	30.3	

It is anticipated that the requirements of the secondary battery will present different technological problems from that of the four-cycle-life battery. Consequently this task is programed for a much longer development time. Item 3) above may be expected to have completed development by mid-1970.

Power requirements imposed on the BVS power subsystem by the sequence of events are shown in table 58 together with a computation of the energy needs. The energy requirements for the initial orbit, composed of the transmission period through entry, a data accumulation period and transmission of these data, are 262.39 W-h. Subsequent orbits composed of a data accumulation period followed by their transmission require 204.97 W-h. A profile for the initial regime is shown in figure 67.

Mechanization Options and Rationale for Baseline Selection: Options which have been previously considered for supplying power to the BVS include use of batteries, radioisotope thermoelectric generators (RTG) and solar cell systems. Mission times with an all-battery system are limited to a few days because the battery is a finite energy source and weight allocations are limited. The RTG and solar cells, being power sources, have the capability of providing for extended periods of operation. Recent studies indicate that light intensities in the Venusian clouds are higher than previously assumed. Confirmation of this finding would give a decided advantage to a solar cell subsystem from the standpoint of specific weight.

Table 58.- SEQUENCE OF BVS POWER REQUIREMENTS, ORBITAL MISSION

Time	Equipment	Load, W	Energy, W-h
E - 5 to St + 1 (07:47) 0.113 hr	Transmitter DHS Accelerometer Programer BVS sequencer	67 10.5 2.1 4 2 85.6	9.68
St + 1 to D - 0 (14:38) 0.244 hr	Transmitter DHS Accelerometer Programer BVS sequencer	67 10.5 2.1 4 2 85.6	20.90
	Radar (5 sec out of 50 sec)	20	.49
D - 0 to D + 2:40 (2:40) 0.0445 hr	Transmitter DHS Bio lab Mass spectrometer Accelerometer Programer BVS sequencer	67 10.5 1 8 2.1 4 2 94.6	4.22
	Radar (5 sec out of 50 sec)	20	.09
D + 2:40 to D + 12:40 (10:00) 0.167 hr	Transmitter DHS Bio lab Programer BVS sequencer	67 10.5 1 4 2 84.5	14.10
D + 12:40 to D + 1 hr (47:20) 0.788 hr	Bio lab Accelerometer Programer	1 2.1 4 7.1	5.60
D + 1:00:00 to D + 1:02:10 (02:10) 0.036 hr (at 1-hr intervals) (multiplier of 20)	DHS Bio lab Science Programer	10.5 1.0 12.3 4 27.8	20.00
	Radar (5 sec out of 50 sec)	20	.56
D + 1:02:10 to D + 2:00:00 (57:15) 0.955 hr (each hour)	Bio lab Programer	1 4 5	95.50
D + 5:40:00 to D + 6:00:00 (20:00) 0.33 hr	Gas chromatograph	5	1.67

TABLE 58.- SEQUENCE OF BVS POWER REQUIREMENTS, ORBITAL MISSION - Concluded

Time	Equipment	Load, W	Energy, W-h
D + 6:00:00 to D + 6:02:40	DHS	10.5	
(02:40)	Programer	4	
0.045 hr	Mass spectrometer	8	
(Also at 12, 18, and 24 hr)	Gas chromatograph	5	
	Science	12.3	
		39.8	7.09
	Radar	20	.11
	(5 sec out of 50 sec)		
D + 6:02:40 to D + 7:00:00	Bio lab	1	
(57:20)	Accelerometer	2.1	
0.956 hr	Programer	4	
(Also after 12, 18, and 24 hr)		7.1	27.20
R - 5:00:00 to R - 0	Receiver		
5.0 hr	Detector	5	25.00
	Decoder		
R - 0 to R + 00:10:00	Transmitter	67	
(10:00)	DHS	10.5	
0.17 hr	Programer	4.0	
	Bio lab	1.0	
	Doppler	1.5	
	Receiver, etc.	5	
		89	15.15
	Radar	20	.8
	(2-5 sec pulses)		
R + 10:00 to R + 1:30:00	Bio lab	1	
(1:20:00)	Doppler	1.5	
1.33 hr	Receiver, etc.	5	
	Programer	4	
		11.5	15.30
Initial orbit			
Listed items			247.54
Losses 6%			14.85 W-h
Total			262.39
Subsequent orbits			
Listed items			193.37
Losses 6%			11.60 W-h
Total			204.97

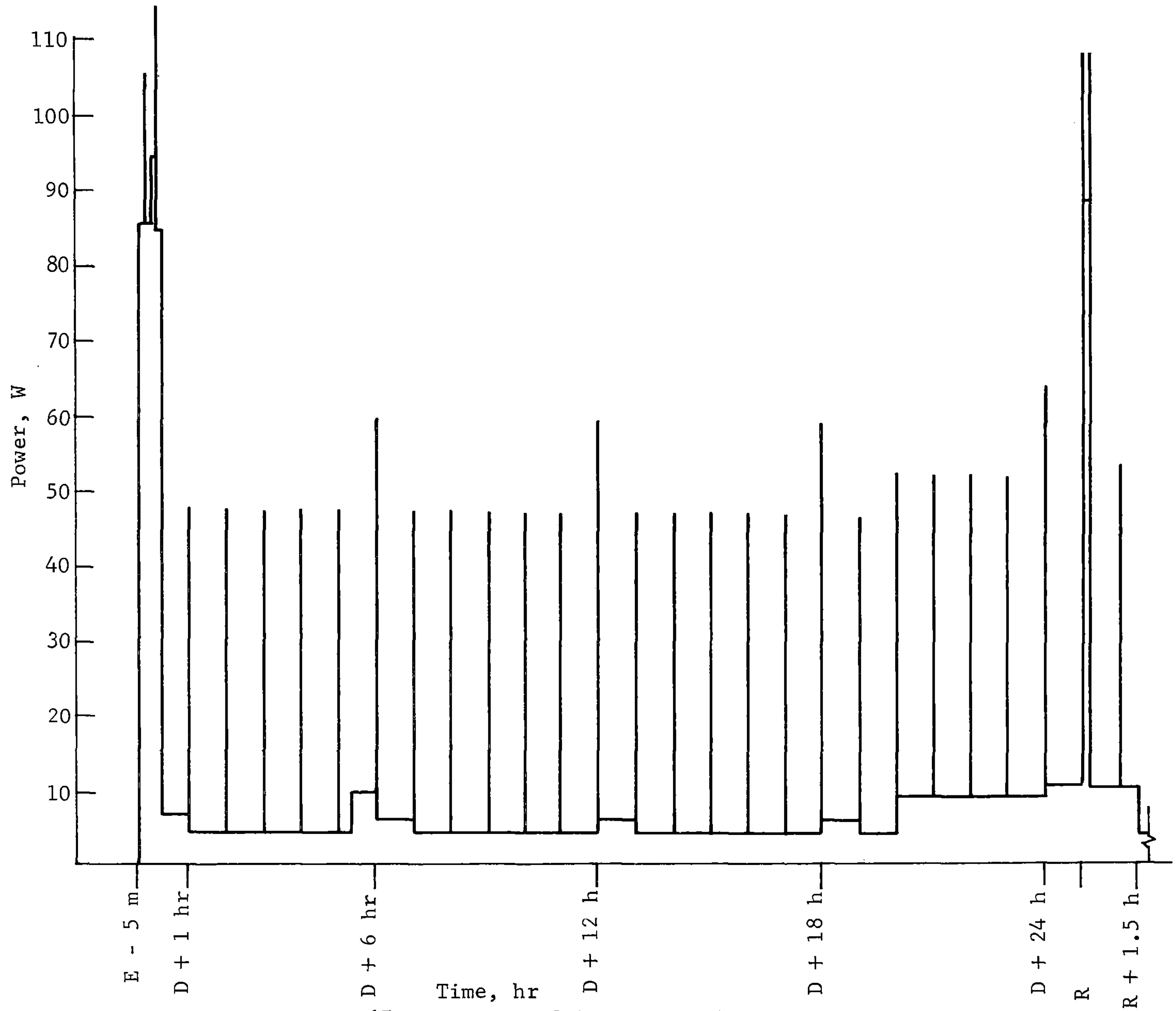


Figure 67.- Power Profile, BVS, Orbital Mission

A net output of 12 W from the solar cell panels will supply the energy requirements for orbits requiring 204.97 W-h. During peak load periods, power is supplied from the battery, otherwise the solar cell panels supply the loads as well as replenish the battery's charge. The solar cell panel is equipped with isolating diodes and wiring, which introduce a 3% loss. The loss in the battery SCR, which switches the battery to the load, is taken at 3% also, while wiring and switching losses in the power control are taken at 6%. During charging the battery is isolated from the bus by the SCR. This reduces the voltage swing on the main bus by allowing the battery discharge voltage to be set near the output voltage of the solar cell panel. It does require, though, that the charger incorporate a voltage stepup feature in its design. Its efficiency is assumed as 0.85. These considerations result in the following values, as taken at the main bus:

1) Energy supplied from battery to bus - 35.2 W-h;
2) Energy supplied to charger - 59.2 W-h;
3) Energy supplied to power control by solar panels - 170.87 W-h;
4) Total energy supplied per orbit - 230.07 W-h.

This quantity of energy can be supplied by the solar panels in:

$$\frac{230.07 \text{ W-h}}{12 \text{ W}} \doteq 19.2 \text{ hr}$$

which results in sufficient margin for a 25-hr orbit.

With an available solar cell area of 11.5 sq ft, a requirement of 12 W would necessitate only an output of 1.04 W/sq ft from the panel. At Earth, an oriented solar panel is capable of 10 W/sq ft. In the clouds of Venus, the light will be diffused and all solar cells will be active. Based on present tentative information on available light intensities, it is expected that operation of the BVS, from the power subsystem standpoint, will be sustained for a substantial portion of the distance from the subsolar point to the terminator.

Aeroshell Power Subsystem for the Orbiter Mission

The arrangement of this subsystem is shown in figure 66 and described in the BVS power subsystem section. The sterilized silver-zinc battery serves as the energy source for both the aeroshell-mounted equipment and for the BVS equipment up to 5 minutes before entry. The battery is charged during transit by equipment mounted on the adapter. A transfer switch, activated by a discrete from the spacecraft before separation, connects the battery to the subsystem.

Table 59 list the sequence of events resulting in power requirements. The energy requirement is 33.56 W-h, and the power profile is shown in figure 68. The items making up the power subsystem, together with their development status, are tabulated (the weights do not include supports or wiring and connector).

Item	Weight, lb	Status
Ag-Zn battery	2.6	Under development
Transfer switch	1.0	Available
Power control module	1.5	Components developed
Isolating diodes	1.0	Available
Total	6.1	

BVS Power Subsystem for the Venus Flyby Mission

The power subsystem for the BVS flyby is similar to the orbiter power subsystem, except that no solar array is used and the battery size is changed to meet the requirements of a different power profile. A block diagram of the subsystem is shown in figure 69. Equipment, estimated weights, and development status are tabulated (the weights do not include supports or wiring and connectors).

Item	Weight, lb	Status
Ag-Zn battery	20.7	Under development
Isolating diode	.2	Available
Power transfer switch	1.0	Available
Power control module	3.0	Current technology
Total	24.9	

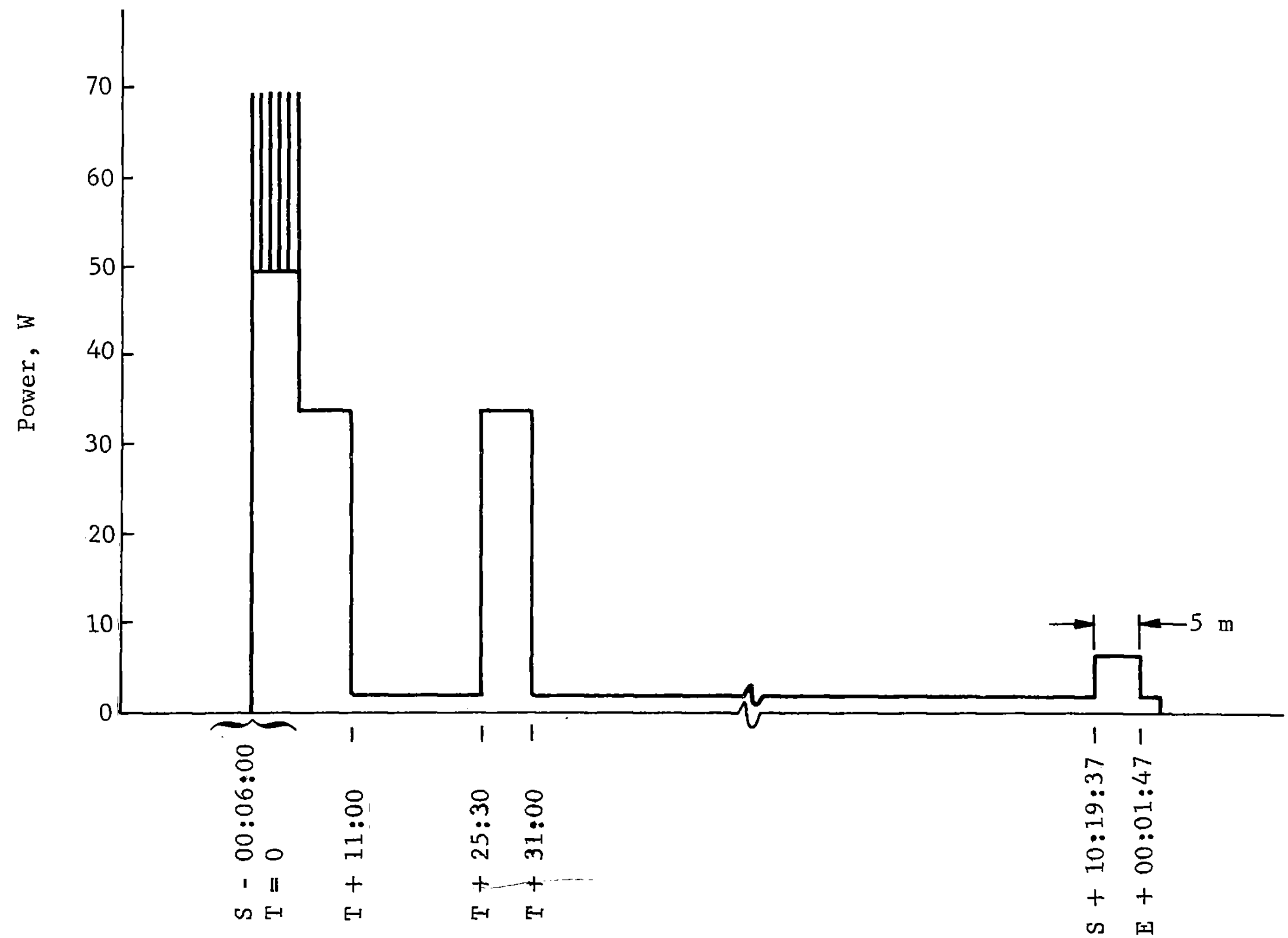


Figure 68.- Power Profile, Aeroshell, Orbital Mission

TABLE 59.- SEQUENCE OF AEROSHELL POWER REQUIREMENTS, ORBITAL MISSION

Time	Equipment	Load, W	Energy, W-h
T - 0 S - 00:06:00 to T + 5:00 (5:00) 0.083 hr	DHS Transmitter Bio lab Mass spectrometer Gas chromatograph Accelerometer Aeroshell Data subsystem	15.5 12 1 8 5 2.1 2 4 49.6	4.14
	Radar (5 sec out of 50 sec)	20	.17
T + 5:00 to T + 11:00 (06:00) 0.1 hr	Data subsystem DHS Transmitter Aeroshell sequencer	4 15.5 12 2 33.5	3.35
T + 11:00 to T + 25:30 (14:30) 0.242 hr	Aeroshell sequencer	2	.48
T + 25:30 to T + 31:00 (05:30) 0.092 hr	Aeroshell sequencer DHS Transmitter Data subsystem	2 15.5 12 4 33.5	3.09
T + 31:00 S + 00:25:00 to S + 10:19:37 (09:54:37) 9.913 hr	Aeroshell sequencer	2	19.83
S + 10:19:37 to S + 10:24:37 (E - 00:00:00) (05:00) 0.083 hr	Instrument power supply Multiplexer/encoder Aeroshell sequencer	0.5 4.0 2.0 6.5	.54
E - 00:00:00 to E + 00:01:47 (01:47) 0.0297 hr	Aeroshell sequencer	2	.06
Listed items			31.66 W-h
Losses, 6%			1.90 W-h
Total			33.56 W-h

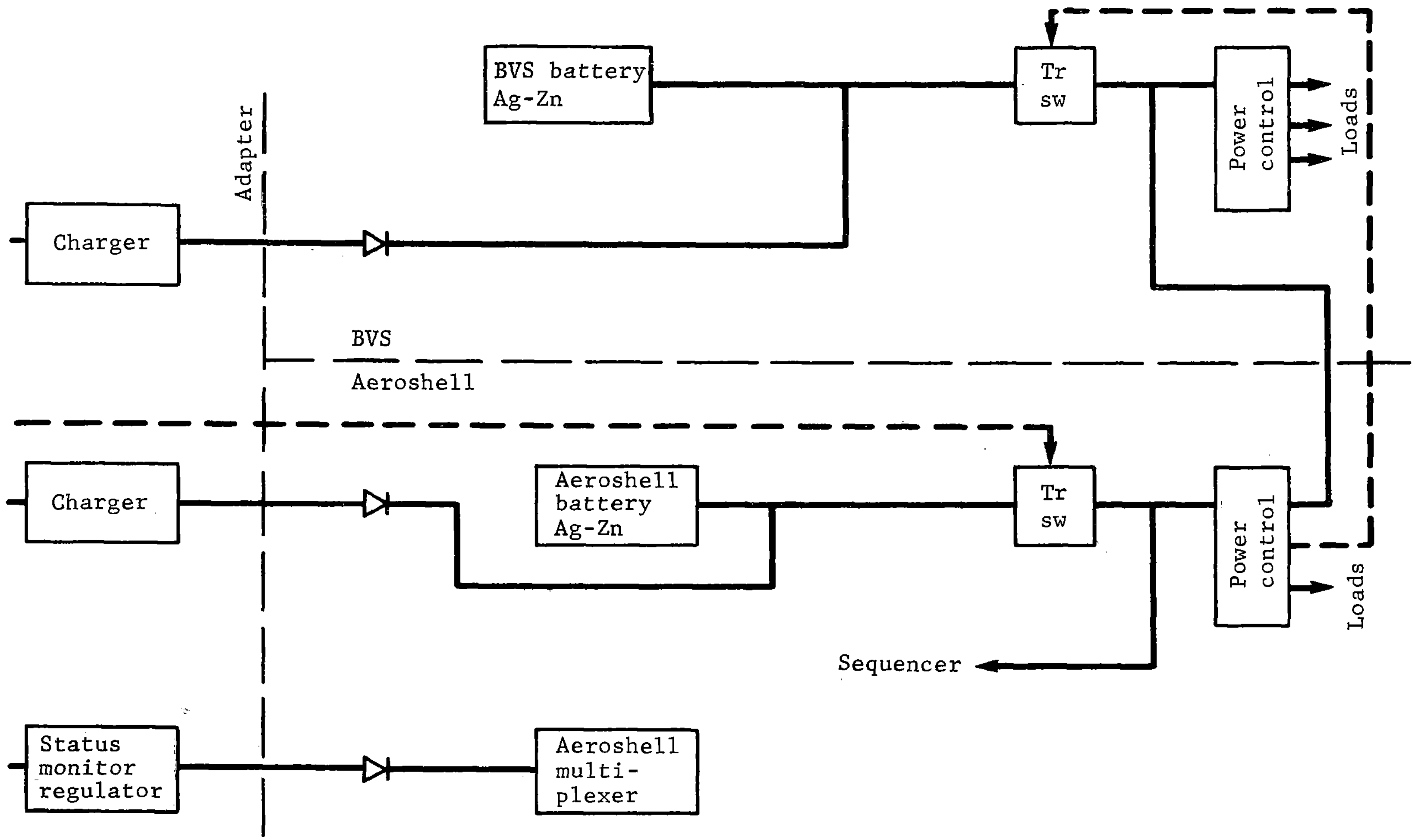


Figure 69.- Power Subsystem

Power requirements imposed on the BVS power subsystem by the sequence of events are shown in table 60 together with a computation of the energy needs. A profile for the initial cycle is shown in figure 70. The energy requirements for a sequence of cycles totaling about 50 hr is tabulated.

Item	Energy, W-h
Initial cycle	135.8
Subsequent cycles (3 at 104.9 W-h)	314.7
Cycles with sonde sequence (2 at 114.8 W-h)	229.6
Final transmission 1.06 (1.8 + 41.6) W-h	46.0
Total	726.1

Aeroshell Power Subsystem for the Venus Flyby Mission

The aeroshell power subsystem for the Venus flyby mission is the same as the orbiter mission except for the power profile that is shown in figure 71. A block diagram is shown in figure 69.

The sequence of events resulting in power requirements is listed in table 61. The resulting energy requirements are 176.5 W-h from the battery.

A listing of the items making up the power subsystem together with development status and weight is tabulated (the weights do not include supports or wiring and connectors).

Item	Weight, lb	Status
Ag-Zn battery	8.4	Under development
Transfer switch	1.0	Available
Power control module	1.5	Components developed
Isolating diodes	.4	Available
Total	11.3	

TABLE 60.- SEQUENCE OF BVS POWER REQUIREMENTS, FLYBY MISSION

Time	Equipment	Load, W	Energy, W-h
E + 00:01:09 to P + 00:01:00	BVS sequencer	2.0	
(00:16:17)	Relay transmitter	67.0	
0.271 hr	DHS	14.5	
	Accelerometer	2.1	
	Pressure sensors	2.8	
	Temperature sensors	2.0	
	H_2O sensor	.4	
	Light backscatter	3.9	
	Solar aspect	1.3	
	Radar altimeter	2.0	
		98.0	26.6
P + 00:01:00 to P + 00:04:00	DHS	14.5	
(00:03:00)	Sensors (as above)	10.4	
0.05 hr	Mass spectrometer	8.0	
	Gas chromatograph	6.0	
	Mini bio lab	1.0	
	Radar altimeter	2.0	
	S-band receiver	2.5	
	S-band modulator/exciter	2.0	
	Command detector and decoder	7.5	
	S-band power amplifier (warmup)	2.0	
		55.9	2.8
P + 00:04:00 to P + 00:24:00	DHS	14.5	
(00:20:00)	Sensors	10.4	
0.333 hr	Gas chromatograph	6.0	
	Mini bio lab	1.0	
	Radar altimeter	2.0	
	S-band receiver	2.5	
	S-band modulator/exciter	2.0	
	Command detector and decoder	7.5	
	Power amplifier	84.0	
		129.9	43.3
P + 00:24:00 to P + 00:24:05	DHS	14.5	
(00:00:05) (Occurs only on initial cycle)	Gas chromatograph	6.0	
	Mini bio lab	1.0	
0.0014 hr		21.5	.03
P + 00:24:05 to P + 01:00:00	DHS (programer only)	4.0	
(00:35:55)	Mini bio lab	1.0	
0.600 hr	Gas chromatograph	1.0	
		6.0	3.6
P + 01:00:00 to P + 01:02:00	DHS	14.5	
(00:02:00)	Sensors	10.4	
0.033 hr	Altimeter	2.0	
	Mini bio lab A	1.0	
	Gas chromatograph	1.0	
	Receiver	2.5	
	Command detector	1.5	
		32.9	1.1
P + 01:02:00 to P + 01:05:00	DHS (programer only)	4.0	
(00:03:00)	Mini bio lab	1.0	
0.05 hr	Gas chromatograph B	1.0	
	Receiver	2.5	
	Command detector	1.5	
		10.0	.5

TABLE 60.- SEQUENCE OF BVS POWER REQUIREMENTS, FLYBY MISSION - Continued

Time	Equipment	Load, W	Energy, W-h
P + 01:05:00 to P + 02:00:00 (00:55:00) 0.915 hr	DHS (programer only) Mini bio lab } C Gas chromatograph	4.0 1.0 1.0 6.0	5.5
P + 02:00:00 to P + 02:02:00	As in A	32.9	1.1
P + 02:02:00 to P + 02:05:00	As in B	10.0	.5
P + 02:05:00 to P + 03:00:00	As in C	6.0	5.5
P + 03:00:00 to P + 03:02:00	As in A	32.9	1.1
P + 03:02:00 to P + 03:05:00	As in B	10.0	.5
P + 03:05:00 to P + 04:00:00	As in C	6.0	5.5
P + 04:00:00 to P + 04:02:00	As in A	32.9	1.1
P + 04:02:00 to P + 04:05:00	As in B	10.0	.5
P + 04:05:00 to P + 05:00:00	As in C	6.0	5.5
P + 05:00:00 to P + 05:02:00	As in A	32.9	1.1
P + 05:02:00 to P + 05:05:00	As in B	10.0	.5
P + 05:05:00 P + 05:40:00 (00:35:00) 0.583 hr	DHS (programer only) Mini bio lab Gas chromatograph	4.0 1.0 1.0 6.0	3.5
P + 05:40:00 to P + 06:00:00 (00:20:00) 0.333 hr	DHS Mini bio lab Gas chromatograph	4.0 1.0 6.0 11.0	3.7
P + 06:00:00 to P + 06:02:00 (00:02:00) 0.033 hr	DHS Sensors Mass spectrometer Gas chromatograph Mini bio lab Radar altimeter Receiver Command detector	14.5 10.4 8.0 6.0 1.0 2.0 2.5 1.5 44.9	1.5
P + 06:02:00 to P + 06:05:00	As in B	10.0	.5
P + 06:05:00 to P + 07:00:00	As in C	6.0	5.5
P + 07:00:00 to P + 07:02:00	As in A	32.9	1.1
P + 07:02:00 to P + 07:05:00	As in B	10.0	.5
P + 07:05:00 to P + 08:00:00	As in C	6.0	5.5
P + 08:00:00 to P + 08:02:30 (00:02:30) 0.0417 hr	DHS Sensors Mass spectrometer Gas chromatograph Mini bio lab Radar altimeter S-band receiver S-band modulator/exciter Command detector and decoder S-band power amplifier (warmup)	14.5 10.4 8.0 1.0 1.0 2.0 2.5 2.0 7.5 2.0 42.9	1.8

TABLE 60.- SEQUENCE OF BVS POWER REQUIREMENTS, FLYBY MISSION - Concluded

Time	Equipment	Load, W	Energy, W-h
P + 08:02:30 to P + 08:22:30	DHS	14.5	
(00:20:00)	Sensors	10.4	
0.333 hr	Altimeter	2.0	
	Mini bio lab	1.0	
	S-band receiver	2.5	
	Command detector	1.5	
	Command decoder	6.0	
	S-band modulator/exciter	2.0	
	S-band power amplifier	84.0	
	Gas chromatograph	1.0	
		124.9	41.6
Initial cycle			
Listed items			128.1
Losses 6%			7.7
Total			135.8
Subsequent cycles			
Listed items			99.0
Losses 6%			5.9
Total			104.9
X + 00:00:00 to X + 00:36:00	For Sonde Sequence		
(36:00)	DHS	13.5	
0.6 hr	Sonde receiver	1.0	
	Bit synchronizer	1.0	
Total		15.5	9.3
	Subsequent cycles including sonde sequence		
	Listed items		108.3
	Losses 6%		6.5
Total			114.8

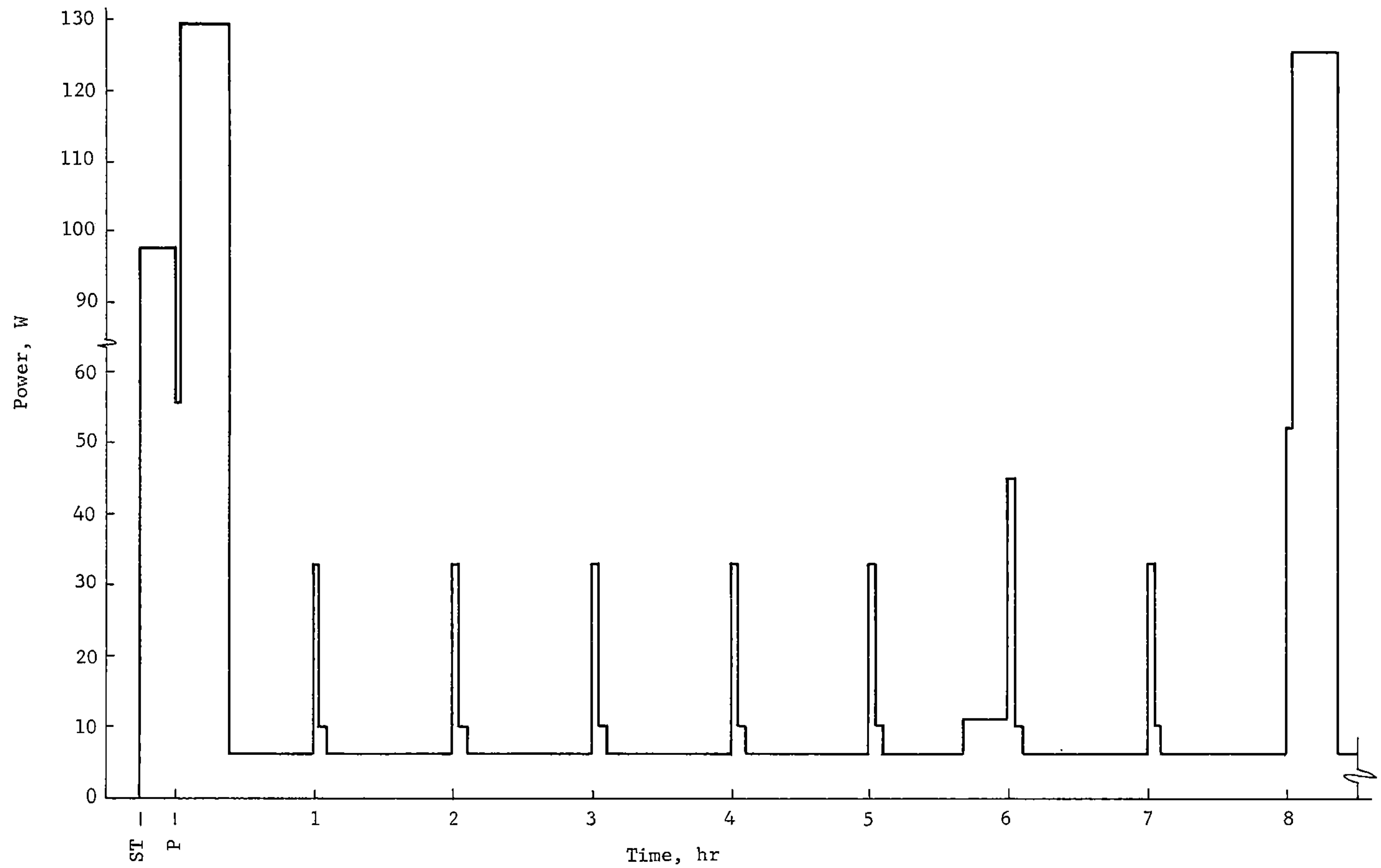


Figure 70.- Power Profile, BVS, Flyby Mission

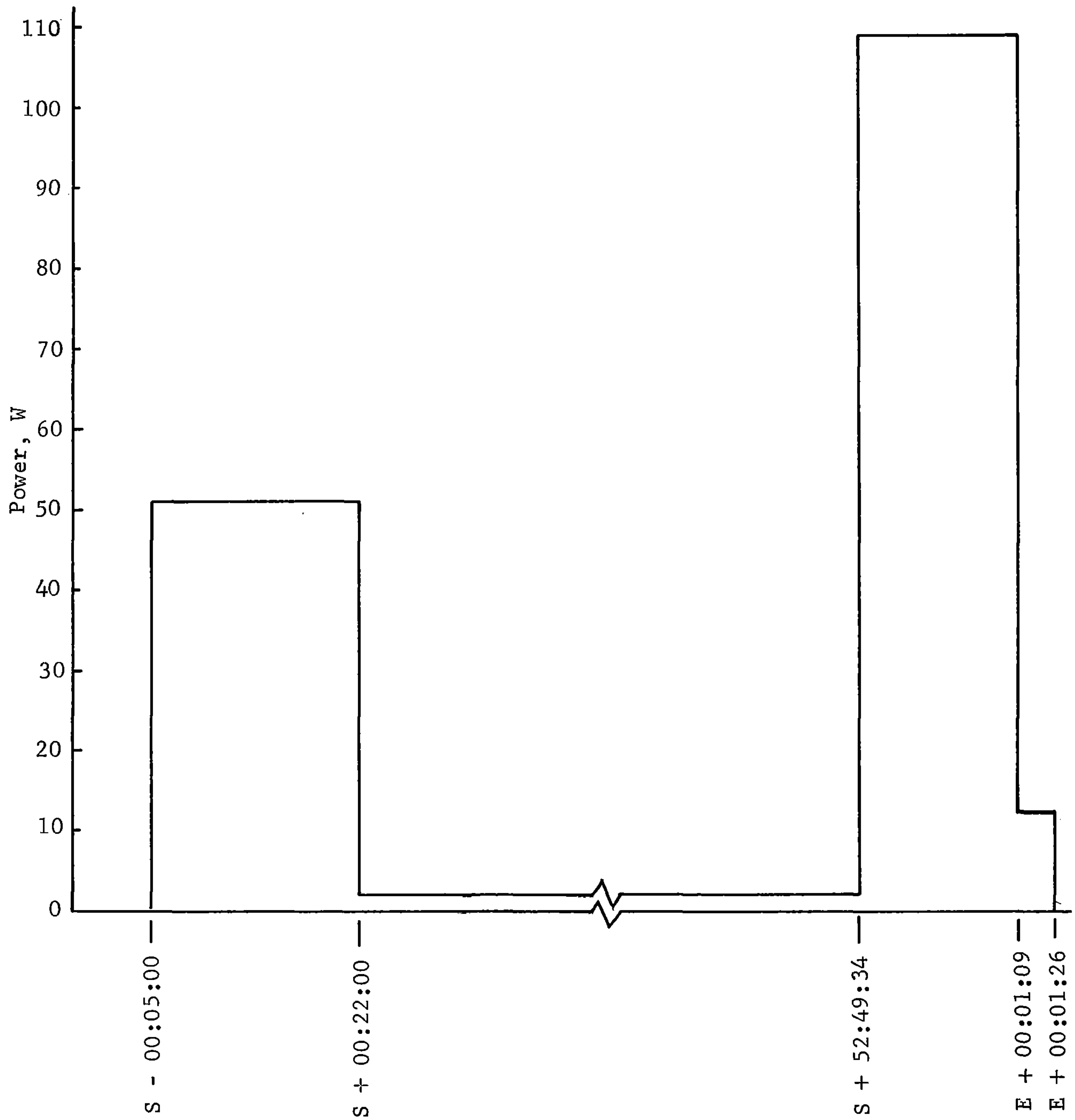


Figure 71.- Power Profile, Aeroshell, Flyby Mission

TABLE 61.- SEQUENCE OF AEROSHELL POWER REQUIREMENTS (FLYBY MISSION)

Time	Equipment	Load, W	Energy, W-h
S - 00:05:00 to S + 00:22:00	Aeroshell sequencer	2.0	
(27:00)	Aeroshell multiplexer/encoder	4.0	
0.45 hr	Aeroshell transducer power supply	1.0	
	Aeroshell science, uv photometers)	5.0	
	BVS temperature-pressure	10.4	
	BVS accelerometer	2.1	
	BVS DHS	14.5	
	BVS relay transmitter	12.0	
		51.0	23.0
S + 00:22:00 to S + 52:49:35	Aeroshell sequencer	2.0	104.9
(52:27:35)			
52.46 hr			
S + 52:49:35 to E + 00:01:09	Aeroshell sequencer	2.0	
(00:21:08)	Aeroshell multiplexer/encoder	4.0	
0.35 hr	Aeroshell transducer power supply	1.0	
	Aeroshell science (uv photometers)	5.0	
	BVS temperature-pressure	10.4	
	BVS accelerometer	2.1	
	BVS DHS	15.5	
	BVS relay transmitter	67.0	
	BVS sequencer	2.1	
		109.1	38.5
E + 00:01:09 to E + 00:01:26	Aeroshell sequencer	2.0	
(00:00:17)	Aeroshell multiplexer/encoder	4.0	
0.0047 hr	Aeroshell transducer power supply	1.0	
	Aeroshell uv photometer	5.0	
		12.0	.057
Listed items			166.5
Losses, 6%			10.0
Total			176.5

BVS Power Subsystem for the Venus/Mercury Flyby Mission

Requirements are similar to those for the Venus flyby mission, except that average power requirements are higher.

The equipment is shown in figure 69. It is the same as that for the Venus flyby mission except for battery capacity. The silver-zinc battery is sized to support a basic 50-hr mission. No option to use solar energy is available for this mission because the station is deployed on the dark side of the planet. Estimates of weight and development status are tabulated.

Item	Weight, lb	Status
Ag-Zn battery	28.2	Under development
Isolating diode	.2	Available
Power transfer switch	1.0	Available
Power control module	3.0	Current technology
Total	32.4	

Power requirements imposed upon the BVS power subsystem by the sequence of events are shown in table 62 together with a computation of the energy needs. A power profile for the initial cycle is shown in figure 72. Wiring and switching losses in the power control are taken as 6% of the delivered power. The total energy requirements for a sequence totaling about 50 hr are tabulated.

Item	Energy, W-h
Initial cycle	146.2
Subsequent cycles (3 at 104.6 W-h)	313.8
Cycles with sonde sequence (2 at 114.5 W-h)	229.0
Final transmission 1.06 (1.8 + 41.6) W-h	46.0
Total	735.0

TABLE 62.- SEQUENCE OF BVS POWER REQUIREMENTS, VENUS/MERCURY

Time	Equipment	Load, W	Energy, W-h
E + 00:01:25 to P + 00:01:00	BVS sequencer	2.0	
(00:16:01)	DHS	14.5	
0.27 hr	Accelerometer	2.1	
	Pressure sensors	2.8	
	Temperature sensors	2.0	
	H_2O sensor	0.4	
	Light backscatter	3.9	
	Solar aspect	1.3	
	Radar altimeter	2.0	
	S-band receiver	2.5	
	S-band mod/exciter	2.0	
	S-band power amplifier	84.0	
	(Occurs only on initial cycle)	119.5	32.3
P + 00:01:00 to P + 00:04:00	DHS	14.5	
(00:03:00)	Sensors (as above)	10.4	
0.05 hr	Mass spectrometer	8.0	
	Gas chromatograph	6.0	
	Mini bio lab	1.0	
	Radar altimeter	2.0	
	S-band receiver	2.5	
	S-band modulator/exciter	2.0	
	Command detector and decoder	7.5	
	S-band power amplifier	84.0	
	(Occurs only on initial cycle)	137.9	6.9
P + 00:04:00 to P + 00:24:00	DHS	14.5	
(00:20:00)	Sensors	10.4	
0.333 hr	Gas chromatograph	6.0	
	Mini bio lab	1.0	
	Radar altimeter	2.0	
	S-band receiver	2.5	
	S-band modulator/exciter	2.0	
	Command detector and decoder	7.5	
	Power amplifier	84.0	
		129.9	43.3
P + 00:24:00 to P + 00:24:05	DHS	14.5	
(00:00:05)	Gas chromatograph	6.0	
0.0014 hr	Mini bio lab	1.0	
	(Occurs only on initial cycle)	21.5	.03
P + 00:24:05 to P + 01:00:00	DHS (programer only)	4.0	
(00:35:55)	Mini bio lab	1.0	
0.600 hr	Gas chromatograph	1.0	
		6.0	3.6
P + 01:00:00 to P + 01:02:00	DHS	14.5	
(00:02:00)	Sensors	10.4	
0.033 hr	Altimeter	2.0	
	Mini bio lab A	1.0	
	Gas chromatograph	1.0	
	Receiver	2.5	
	Command detector	1.5	
		32.9	1.1
P + 01:02:00 to P + 01:05:00	DHS (programer only)	4.0	
(00:03:00)	Mini bio lab B	1.0	
0.05 hr	Gas chromatograph	1.0	
	Receiver	2.5	
	Command detector	1.5	
		10.0	.5

TABLE 62.- SEQUENCE OF BVS POWER REQUIREMENTS, VENUS/MERCURY - Continued

Time	Equipment	Load, W	Energy, W-h
P + 01:05:00 to P + 02:00:00 (00:55:00) 0.915 hr	DHS (programer only) Mini bio lab Gas chromatograph } C	4.0 1.0 1.0 6.0	5.5
P + 02:00:00 to P + 02:02:00	As in A	32.9	1.1
P + 02:02:00 to P + 02:05:00	As in B	10.0	.5
P + 02:05:00 to P + 03:00:00	As in C	6.0	5.5
P + 03:00:00 to P + 03:02:00	As in A	32.9	1.1
P + 03:02:00 to P + 03:05:00	As in B	10.0	.5
P + 03:05:00 to P + 04:00:00	As in C	6.0	5.5
P + 04:00:00 to P + 04:02:00	As in A	32.9	1.1
P + 04:02:00 to P + 04:05:00	As in B	10.0	.5
P + 04:05:00 to P + 05:00:00	As in C	6.0	5.5
P + 05:00:00 to P + 05:02:00	As in A	32.9	1.1
P + 05:02:00 to P + 05:05:00	As in B	10.0	.5
P + 05:05:00 to P + 05:40:00 (00:35:00) 0.583 hr	DHS (programer only) Mini bio lab Gas chromatograph	4.0 1.0 1.0 6.0	3.5
P + 05:40:00 to P + 06:00:00 (00:20:00) 0.333 hr	DHS Mini bio lab Gas chromatograph	4.0 1.0 6.0 11.0	3.7
P + 06:00:00 to P + 06:02:00 (00:02:00) 0.033 hr	DHS Sensors Mass spectrometer Gas chromatograph Mini bio lab Radar altimeter Receiver Command detector	14.5 10.4 8.0 6.0 1.0 2.0 2.5 1.5 45.9	1.5
P + 06:02:00 to P + 06:05:00	As in B	10.0	.5
P + 06:05:00 to P + 07:00:00	As in C	6.0	5.5
P + 07:00:00 to P + 07:02:00	As in A	32.9	1.1
P + 07:02:00 to P + 07:05:00	As in B	10.0	.5
P + 07:05:00 to P + 08:00:00	As in C	6.0	5.5
P + 08:00:00 to P + 08:02:30 (00:02:30) 0.0417 hr	DHS Sensors Gas chromatograph Mini bio lab Radar altimeter S-band receiver S-band modulator/exciter Command detector and decoder S-band power amplifier (warmup) (Occurs only on final cycle)	14.5 10.4 1.0 1.0 2.0 2.5 2.0 7.5 2.0 42.9	1.8

TABLE 62.- SEQUENCE OF BVS POWER REQUIREMENTS, VENUS/MERCURY - Concluded

Time	Equipment	Load, W	Energy, W-h
P + 08:02:30 to P + 08:22:30 (00:20:00) 0.333 hr	DHS	14.5	
	Sensors	10.4	
	Altimeter	2.0	
	Mini bio lab	1.0	
	S-band receiver	2.5	
	Command detector	1.5	
	Command decoder	6.0	
	S-band modulator/exciter	2.0	
	S-band power amplifier	84.0	
	Gas chromatograph	1.0	
	(Occurs only on final cycle)	124.9	41.6
	Initial cycle		
	Listed items		137.9
	Losses 6%		8.3
	Total		146.2
	Subsequent cycles		
	Listed items		98.7
	Losses 6%		5.9
			104.6
X + 00:00:00 to X + 00:36:00 (36:00) 0.6 hr	For sonde sequence		
	DHS	13.5	
	Sonde receiver	1.0	
	Bit synchronizer	1.0	
	Total	15.5	9.3
	Subsequent cycles including sonde sequence		
	Listed items		108.0
	Losses 6%		6.5
	Total		114.5

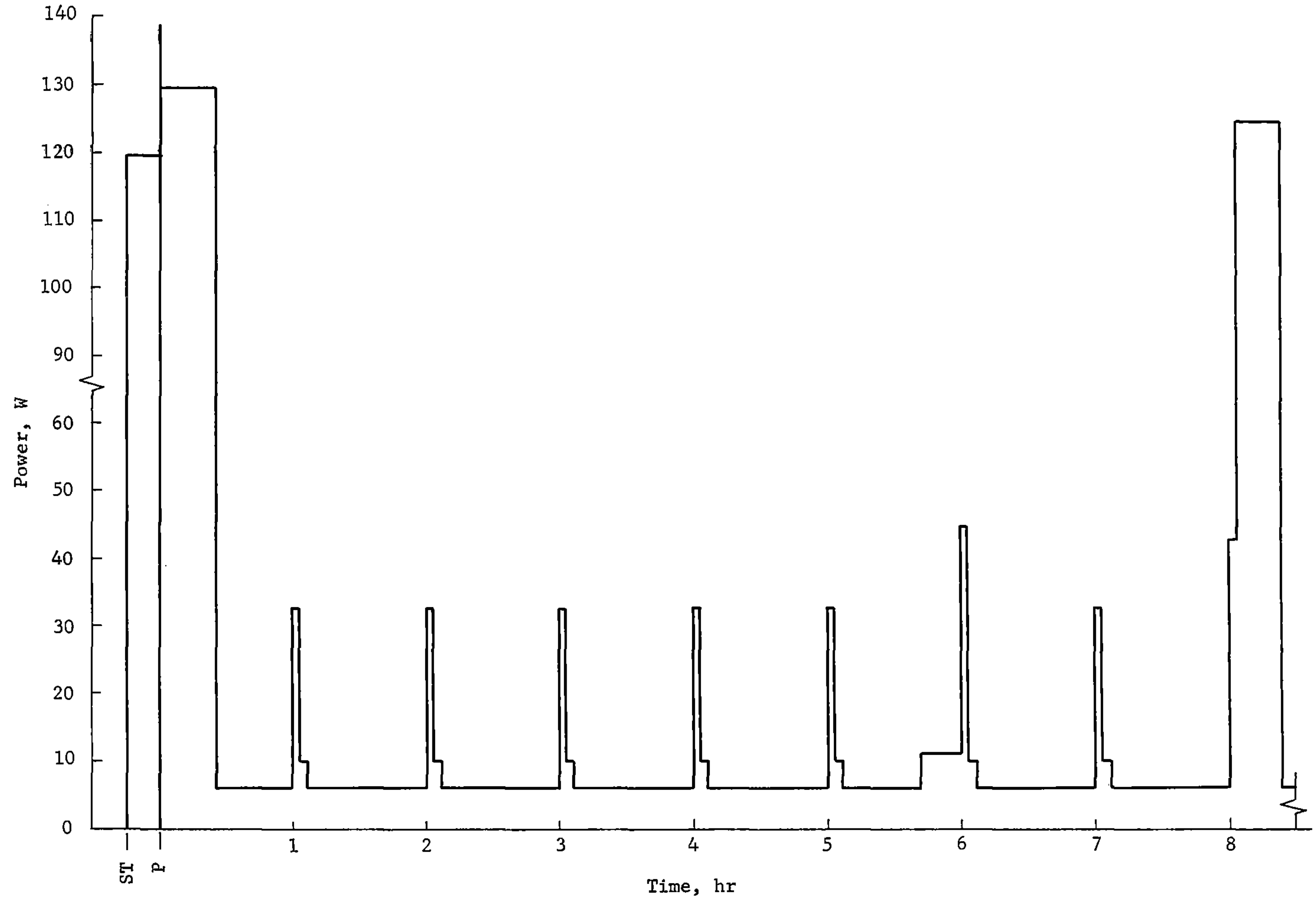


Figure 72.- Power Profile, BVS, Venus/Mercury Flyby Mission, Initial Cycle

Aeroshell Power Subsystem

The block diagram for this subsystem is shown in figure 69. The subsystem is the same as the subsystem for the flyby mission except for the power profile.

The sequence of events resulting in power requirements is listed in table 63. This results in energy requirements of 254.8 W-h from the battery.

The resulting power profile is shown in figure 73. A listing of the items making up the power subsystem together with development status is tabulated.

Item	Weight, lb	Status
Ag-Zn battery	11.3	Under development
Transfer switch	1.0	Available
Power control module	1.5	Components developed
Isolating diodes	.4	Available
Total	14.2	

BVS Pyrotechnics for the Orbital Mission

Pyrotechnic assemblies are required for the BVS gondola and the inflation system. Capacitors are used for energy storage and solid-state switches for safe/arm and firing functions. Requirements for the pyrotechnic subsystem are listed in table 64.

A schematic (typical for the pyrotechnic subsystem) is shown in figure 74. Power is received from the power subsystem, and control signals are received from the sequencing subsystem. Capacitor assemblies provide the energy storage for firing bridgewires and charge current limiting resistors restrict the instantaneous load on the power bus. Safe/arm switches are sequenced so that no functions are armed for more than 1 minute before firing. Up to four squib firing circuits may be used with one safe/arm switch. After all pyrotechnic functions in an event are fired, the switches are reset to the safe position, thus opening the power circuit and removing any load caused by a bridgewire short.

TABLE 63.- SEQUENCE OF AEROSHELL POWER REQUIREMENTS

Time	Equipment	Load, W	Energy, W-h
	A		
S - 00:05:00 to S + 00:13:00	Aeroshell sequencer	2.0	
(00:10:00)	Aeroshell multiplexer/encoder	4.0	
0.3 hr	Aeroshell transducer supply	1.0	
	Aeroshell science (uv photometer)	5.0	
	BVS temperature/pressure	10.4	
	BVS accelerometer	2.1	
	BVS DAS	14.5	
		39.0	11.7
S + 00:00:01 to S + 00:14:58	ACS	26.5	6.6
(00:14:57)			
0.25 hr			
	B		
S + 00:13:00 to S + 00:15:00	Same as A	39.0	
(00:02:00)	Receiver	2.5	
0.03 hr	S-band modulator/exciter	2.0	
	S-band power amplifier (warmup)	2.0	
		45.5	1.4
	C		
S + 00:15:00 to S + 00:35:00	Same as B	45.5	
(00:20:00)	S-band power amplifier	82.0	
0.333 hr		127.5	42.5
S + 00:35:00 to S + 67:53:00	Aeroshell sequencer	2.0	134.6
(67:18:00)			
67.3 hr			
S + 67:53:00 to S + 67:55:00	Same as B	45.5	1.4
(00:02:00)			
0.03 hr			
S + 67:55:00 to S + 68:14:45	Same as C	127.5	42.2
(E + 00:01:25)			
(00:19:45)			
0.33 hr			
E + 00:01:25 to E + 00:01:26	Aeroshell sequencer	2.0	
(00:00:01)	Aeroshell multiplexer/encoder	4.0	
0.0 hr	Aeroshell transducer supply	1.0	
	Aeroshell uv photometer	5.0	
		12.0	
	Listed items		240.4
	Losses, 6%		14.4
	Total		254.8

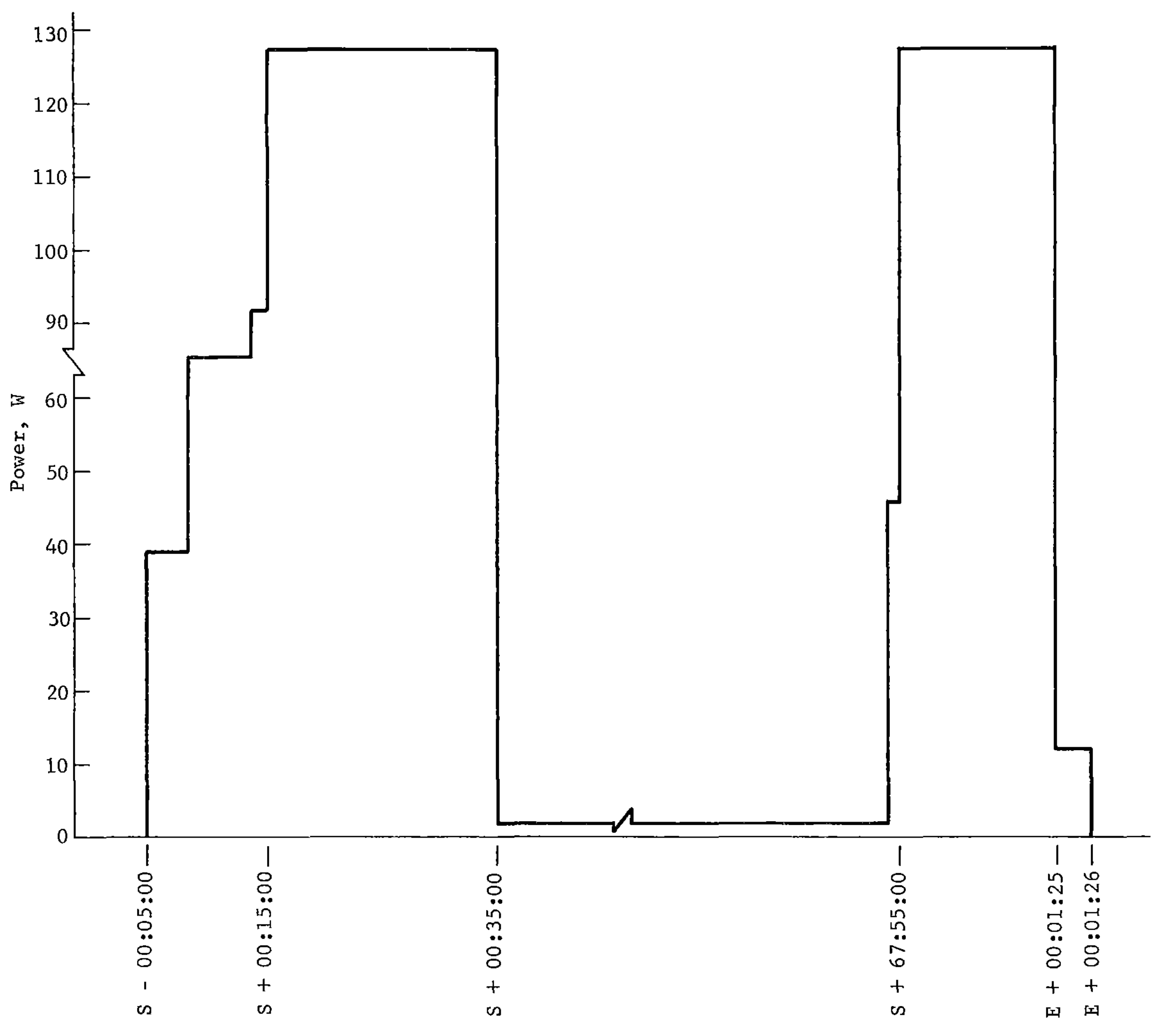


Figure 73.- Power Profile, Venus/Mercury Flyby Mission, Aeroshell

TABLE 64.- PYROTECHNIC REQUIREMENTS

Squib-firing harness connecting the devices to the firing unit must meet the following requirements: 1) Continuous circumferential shielding up to the EED device; 2) Conductors carrying squib-firing current to be physically isolated from other conductors; 3) Radiated power pickup on squib firing lines to be reduced by twisting and shielding the conductors; radiation pickup considerations include bridge-to-bridge and bridge-to-case modes as well as the normal firing mode.
Components and design techniques to be within state-of-the-art technology and proven through use.
Complete testability before being committed to launch.
Squibs will not fire if 1 A/1 W is simultaneously applied to each bridgewire circuit.
Electroexplosive devices will not degrade or be fired when exposed to an rf field intensity of 100 W/m^2.
Two bridgewires used for each function.
Pyrotechnic subsystem firing power to be controlled by a safe/arm device.
Redundant arming and firing circuitry to be used.
No single or common failure mode (including procedural deviation) shall both arm and command the pyrotechnic subsystem.
Use separate energy sources (capacitors) for pyrotechnic firing isolated from other subsystem uses.
Firing energy to be a minimum of 0.150 joules per bridgewire.
12 sec minimum between reuse of firing energy storage capacitors.
Components and assemblies shall withstand dry heat sterilization.
The capsule component temperatures will be controlled within the following limits: Operating: 15 to 140°F Non operating: -35 to 160°F

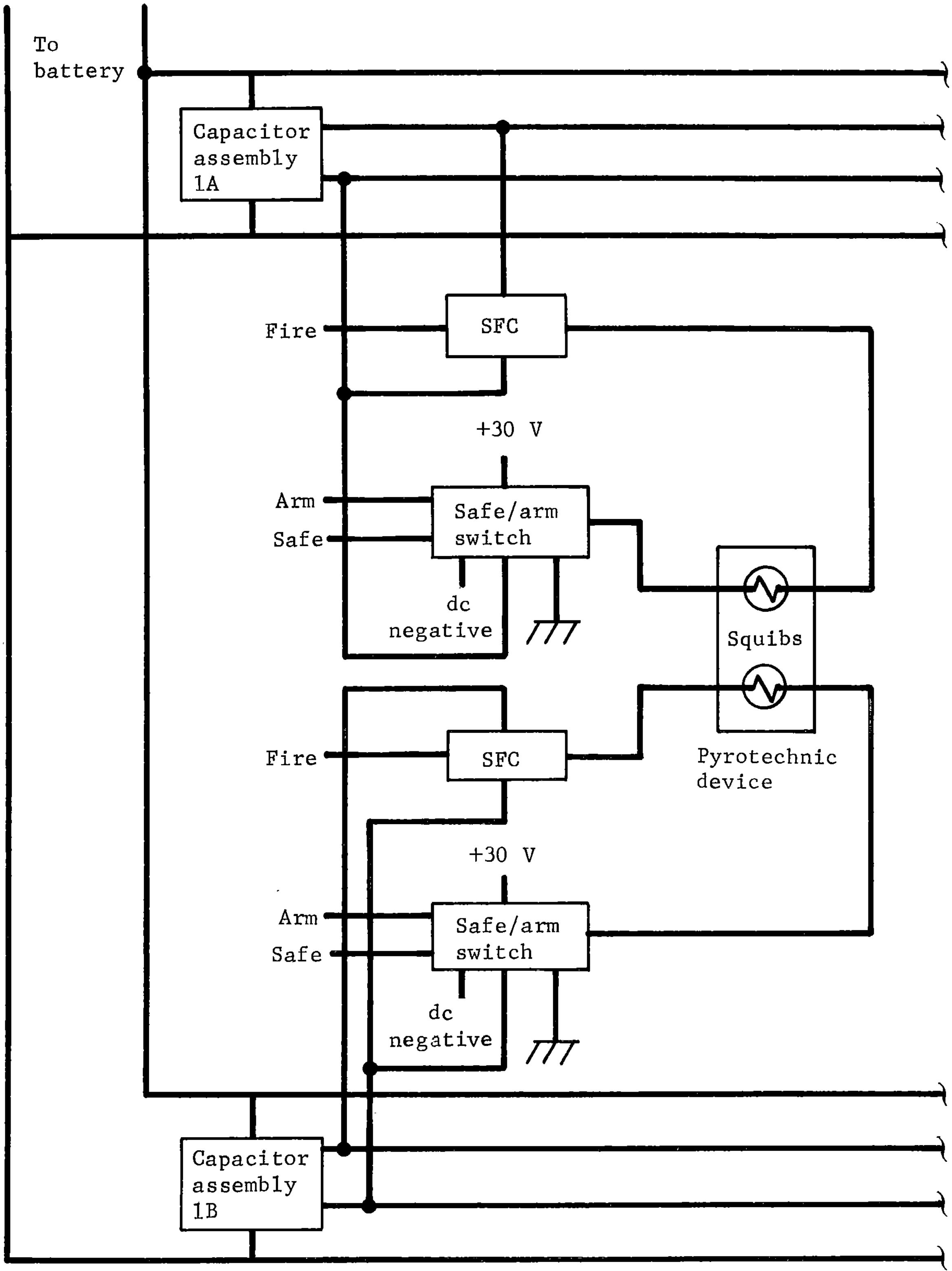


Figure 74.- Typical Pyrotechnic Schematic

The diagram in figure 74 shows the typical redundancy provided for each function. Parallel circuits are provided from the power subsystem through redundant capacitor assemblies, safe/arm switches, and squib firing circuits to one of two squibs in each pyrotechnic device. Generally, two squibs with one bridgewire each are used for each function. With this arrangement, the proper functioning of either circuit branch will fire all associated pyrotechnic devices.

The capacitor assembly charging is initiated at power transfer and, approximately 12 sec later, capacitors are ready for use. A minimum of 12 sec between subsequent events allows the capacitor assemblies to be recharged and used again. Each capacitor assembly provides the required energy to fire one bridgewire in a given event. The events, functions, and required number of bridgewires are tabulated for the pyrotechnic subsystems in tables 65 and 66. For the inflation subsystem, event 2 will establish the number of assemblies at 6, the firings occurring at intervals of 2 sec.

The use of an individual capacitor assembly to fire each bridgewire, rather than the use of a common capacitor bank eliminates the need for current limiting resistors in each bridgewire circuit.

Subsystem weights.- A listing of components and associated weights for each of the pyrotechnic subsystems is given in table 67. In addition to these systems, a cable cutter is needed to release the parachute from the balloon. To avoid long wire runs through the balloon, the capacitors will be charged before balloon inflation. A timer will provide the delay for release. Weights of these items mounted on the balloon canister cover are tabulated.

Item	Quantity	Weight each, lb	Total weight, lb
Capacitor assembly	2	0.16	0.32
Safe/arm switch	2	0.23	0.46
Squib fire switch	2	0.06	0.12
Timer	1	0.50	0.50
Total			1.40

TABLE 65.- SEQUENCE OF EVENTS FOR PYROTECHNIC FUNCTIONS ON INFLATION SYSTEM (TANK TRUSS) ORBITAL MISSION

Event	Function	Time (Typical), hr: min: sec	Bridgewires
1	Separate subsonic probe	St + 00:00:15	4
2	Extract balloon	PP + 00:00:02	2
	Inflate balloon		
	Fire first igniter	PP + 00:00:03	2
	Fire second igniter	PP + 00:00:05	2
3	Terminate inflation	PP + 00:00:58	2
4	Release inflation system	PP + 00:00:60	4

TABLE 66.- SEQUENCE OF EVENTS FOR PYROTECHNIC FUNCTIONS ON BVS GONDOLA FOR ORBITAL MISSION

Event	Function	Time, hr: min: sec	Bridgewires
1	Release instrument reel	PP + 00:01:03	2
2	Release drop sonde	Command	2
3	Release drop sonde	Command	2

TABLE 67.- BVS PYROTECHNIC SUBSYSTEM WEIGHT, ORBITAL MISSION

Item	Quantity	Weight each, lb	Total weight, lb
Inflation subsystem			
Capacitor assembly	6	0.16	0.96
Safe/arm switch	12	0.23	2.76
Squib fire switch	16	0.06	0.96
Subtotal			4.68
Internal cabling and connectors			0.43
Packaging			1.18
Total			6.29
Gondola			
Capacitor assembly	2	0.16	0.32
Safe/arm switch	2	0.23	0.46
Squib fire switch	6	0.06	0.36
Subtotal			1.14
Internal cabling and connectors			0.10
Packaging			0.30
Total			1.54

Aeroshell Pyrotechnics for the Orbital Mission

The aeroshell pyrotechnic subsystem is identical to the BVS subsystem in concept. The events, functions, and required number of bridgewires, however, are unique and are listed in table 68. Event 2 will establish the number of capacitor assemblies required. Because some firings are 1 sec apart, insufficient time is available to allow for recharge, therefore, a total of 12 capacitor assemblies are needed. The equipment required and estimated weights are tabulated.

Item	Quantity	Unit weight, lb	Weight, lb
Capacitor assembly	12	0.16	1.92
Safe/arm switch	20	0.23	4.60
Squib fire switch	48	0.06	2.88
Subtotal			9.40
Internal cabling and connectors			.92
Packaging			3.28
Total			13.60

BVS Pyrotechnic Subsystem for the Venus Flyby Mission

Requirements for the BVS pyrotechnic subsystem are the same as those for the orbital mission except for an increased quantity of components.

The events, functions and required number of bridgewires are tabulated for the gondola and inflation subsystems in tables 69 and 70. After inflation the tanks are dropped from the gondola. For the inflation subsystem, event 2 will establish the number of capacitor assemblies at 8, the firings occurring at intervals of 1 to 2 sec.

Subsystem weights.- A listing of components and associated weights for each of the pyrotechnic subsystems is given in table 71. In addition to these systems, a cable cutter and timer are needed to release the parachute from the balloon. Weights of these items mounted on the balloon canister cover total 1.4 lb as for the orbital mission.

TABLE 68.- SEQUENCE OF EVENTS FOR PYROTECHNIC FUNCTIONS ON AEROSHELL, ORBITAL MISSION

Event	Function	Time, hr: min: sec	Bridgewires
1	Separate forward bio canister	S - 00:04:00	6
2	Separate capsule spinup capsule	S + 00:00:00 S + 00:00:01	8 4
3	Fire bipropellant module	S + 00:20:00	4
4	Shut down propulsion	S + 00:24:17	4
5	Separate propulsion module	S + 00:24:30	6
6	Fire for partial despin	S + 10:19:39	2
7	Deploy afterbody parachute	E + 00:01:15	2
8	Release afterbody	E + 00:01:31	6
9	Release BVS and subsonic probe from aeroshell	E + 00:01:47	6

TABLE 69.- SEQUENCE OF EVENTS FOR PYROTECHNIC SUBSYSTEM ON BVS, VENUS FLYBY

Event	Function	Time, hr: min: sec	Bridgewires
1	Deploy BVS science	P + 00:03:30	2
2	Release drop sonde	Command	2
3	Release drop sonde	Command	2

TABLE 70.- SEQUENCE OF EVENTS FOR PYROTECHNIC SUBSYSTEM ON INFLATION SYSTEM, VENUS FLYBY

Event	Function	Time, hr: min: sec	Bridgewires
1	Separate subsonic probe	St + 00:00:15	4
2	Extract balloon Inflate sock Inflate balloon	P + 00:00:02 P + 00:00:03 P + 00:00:05	4 2 2
3	Release BVS chute	P + 00:00:27	2
4	Terminate inflation	P + 00:00:59	2
5	Release inflation system	P + 00:01:00	6
	Total		22

TABLE 71.- BVS PYROTECHNIC SUBSYSTEM WEIGHT, VENUS FLYBY

Item	Quantity	Unit weight, lb	Weight, lb
Inflation subsystem			
Capacitor assembly	8	0.16	1.28
Safe/arm switch	10	0.23	2.30
Squib fire switch	22	0.06	1.32
Subtotal			4.90
Internal cabling and connections			.49
Packaging			1.71
Total			7.10
Gondola			
Capacitor assembly	2	0.16	.32
Safe/arm switch	2	0.23	.46
Squib fire switch	6	0.06	.36
Subtotal			1.14
Internal cabling and connections			.10
Packaging			.30
Total			1.54

Aeroshell Pyrotechnics for the Venus Flyby Mission

Requirements for the aeroshell pyrotechnic subsystem are the same as those of the orbital mission with the exception of event timing and number of bridgewires.

The events, functions, and required number of bridgewires for the pyrotechnic subsystems are listed in table 72. Event 2 establishes the number of capacitor assemblies needed to be 12, because firings are 1 sec apart.

The equipment required with associated weights are:

Item	Quantity	Unit weight, lb	Weight, lb
Capacitor assembly	12	0.16	1.92
Safe/arm switch	18	0.23	4.14
Squib fire switch	42	0.06	2.52
Subtotal			8.58
Internal cabling and connections			.86
Packaging			3.00
Total			12.44

TABLE 72.- SEQUENCE OF EVENTS FOR PYROTECHNIC SUBSYSTEM ON AEROSHELL, VENUS FLYBY

Event	Function	Time, hr: min: sec	Bridgewires
1	Separate forward bio canister	S - 00:04:39	6
2	Separate capsule	S - 00:00:00	8
	Spinup capsule	S + 00:00:01	4
3	Pressurize propulsion	S + 00:19:50	2
	Initiate propulsion start	S + 00:20:00	2
4	Shut down propulsion	S + 00:21:34	2
5	Separate propulsion module	S + 00:21:39	4
6	Fire for partial despin	S + 52:54:34	2
7	Fire afterbody chute mortar	E + 00:00:56	2
8	Separate afterbody	E + 00:01:10	6
9	Separate BVS	E + 00:01:26 St - 00:00:00	4
	Total		42

BVS Pyrotechnics for the Venus/Mercury Mission

The BVS pyrotechnic subsystem for the Venus/Mercury mission is essentially identical to that of the Venus flyby mission.

Aeroshell Pyrotechnics for the Venus/Mercury Mission

The Venus/Mercury pyrotechnic subsystem requirements and configuration are the same as those for the orbital mission except for the number of components required.

The events, functions, and required number of bridgewires for the pyrotechnic subsystems are listed in table 73. Event 2 establishes the number of capacitor assemblies needed as 10.

TABLE 73.- SEQUENCE OF EVENTS FOR PYROTECHNIC SUBSYSTEM ON AERO-SHELL, VENUS/MERCURY MISSION

Event	Function	Time, hr: min: sec	Bridgewires
1	Separate forward bio canister	S - 00:04:39	6
2	Separate capsule	S - 00:00:00	8
	Initiate capsule ACS	S + 00:00:01	2
3	Pressurize deflection propulsion	S + 00:07:50	2
	Initiate propulsion start	S + 00:08:00	2
4	Shut down propulsion	S + 00:13:07	2
5	Separate propulsion module	S + 00:13:12	4
6	Terminate capsule ACS	S + 00:14:58	2
7	Fire for partial despin	S + 67:57:59	2
8	Deploy afterbody/BVS chute	E + 00:01:10	6
9	Separate BVS and initiate truss sequencer	E + 00:01:26 St + 00:00:00	4
	Total		40

The equipment required with associated weights are as tabulated.

Item	Quantity	Unit weight, lb	Weight, lb
Capacitor assembly	10	0.16	1.60
Safe/arm switch	18	0.23	4.14
Squib fire switch	40	0.06	2.40
Subtotal			8.14
Internal cabling and connections			.86
Packaging			3.00
Total			12.00

FLOTATION SYSTEM

The requirements for the flotation system are essentially identical for the three mission modes under consideration. All mission modes call for a balloon of 18 ft in diameter designed to support a suspended payload (gondola) of approximately 175 lb at a nominal radius of 6108 km. The balloon is deployed and inflated at a nominal dynamic pressure of 1 psf. For two missions, the entry point is on the sunlit side of the terminator; however, the wind pattern model indicates the balloon will probably cross the terminator during its lifetime. The Venus/Mercury mission requires a dark side entry and flotation.

Because of this similarity, a single baseline configuration is described for the flotation system. The concept of a dual-altitude mission is discussed in Appendix E.

The flotation system consists of the parachute and its associated hardware, the balloon, balloon canister, balloon controls, and inflation subsystem.

The concept of a flotation system for a Venus scientific station is feasible; however, three basic supporting research and technology programs are required to establish system design. These include a materials test and balloon fabrication program, an inflation and deployment wind tunnel test program, and an air drop system proof test program.

A balloon dynamics program has been used to study the time histories of the parachute descent trajectory, the balloon inflation process, the balloon flight recovery and stabilization, the gas temperature and venting function. The program is presented by a block diagram in figure 75. The program is basically made up of two parts -- the balloon thermodynamics and the balloon motion dynamics. The heat flux inputs from the atmosphere along with atmospheric properties define the balloon lift that largely governs balloon motion.

The system motion is initiated while the entry capsule is descending on the main parachute. Provision is made for releasing the aeroshell and subsonic probe at the appropriate Mach numbers while continuing the parachute descent. At the desired atmospheric pressure, balloon inflation is initiated by blowing down the inflation gas tanks through an orifice sized to inflate the balloon in a given period of time. The model drops the inflation tanks and parachute simultaneously, a predetermined number of seconds after balloon inflation begins.

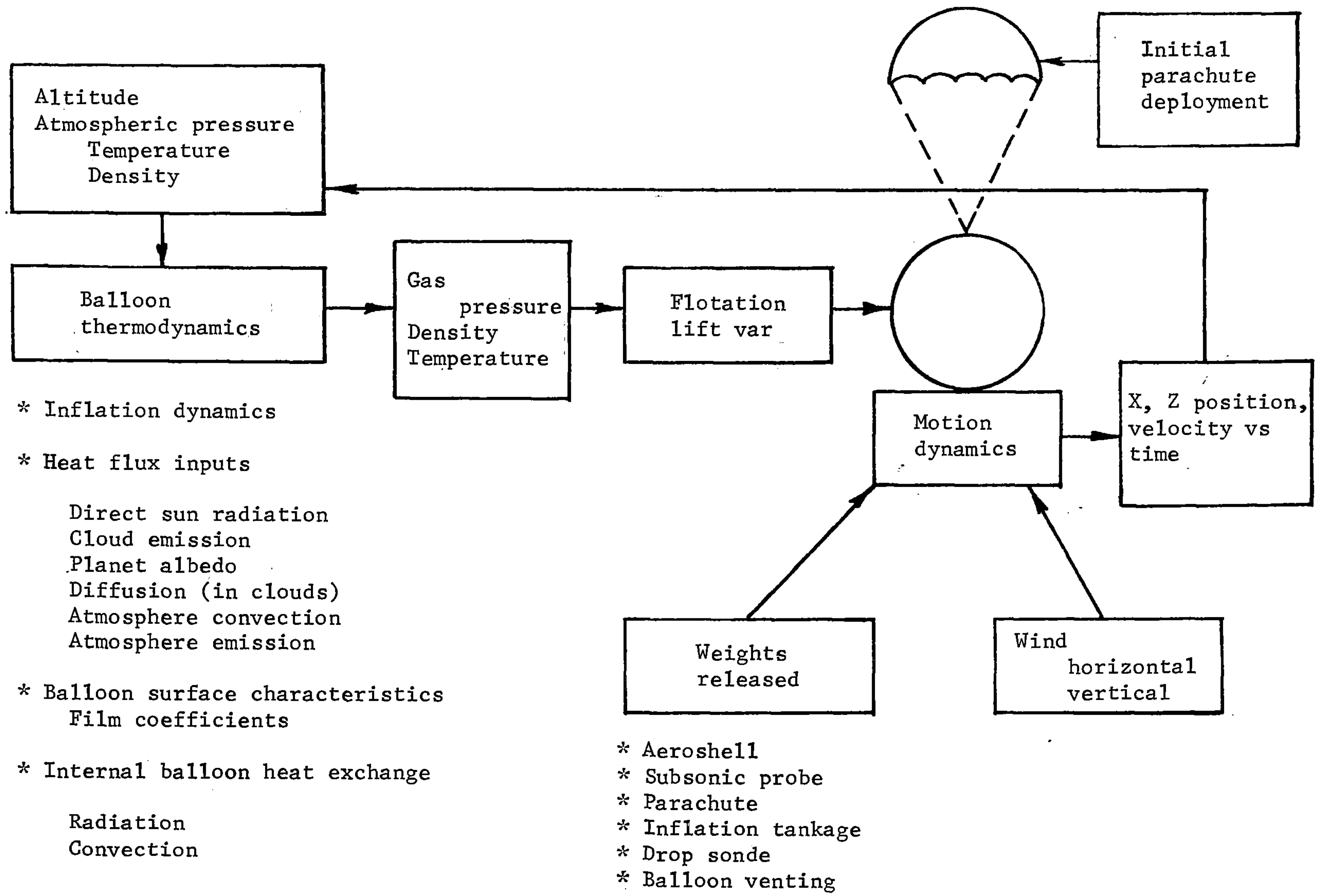


Figure 75.- Balloon Deployment Dynamics Model

During the blowdown period, nominally 45 sec duration, the temperature of the gas expanding into the balloon drops approximately 130°F below ambient. Heat transfer from the atmosphere raises the balloon temperatures to near ambient. Heat transfer from the atmosphere to the balloon gas due to atmospheric radiation, conduction, and convection is simulated for either above or below cloud level operation. The equilibrium float phase is capable of being cloud level operation and is capable of being disturbed by dropping an instrumented sonde or by vertical and horizontal wind perturbations. A venting system is included which operates when the balloon superpressure is exceeded.

The assumptions used for the model are:

1) The balloon with its associated equipment is treated as a rigid body with a point mass;
2) The balloon is assumed to be a sphere with varying diameter during inflation and float phases;
3) The drag forces on the chute, balloon, and aeroshell always line up with the relative wind;
4) The flightpath angle of the entry vehicle in the Venus atmosphere is nearly 90° below the local horizontal by the time balloon inflation begins. The system is assumed to remain in a vertical attitude during the balloon phases;
5) A flat, nonrotating planet with gravitational acceleration constant with altitude is assumed;
6) The balloon is divided into four nodes for heating evaluation. The gas inside the balloon is assumed to be completely homogeneous as to temperature except for the film layer next to the balloon skin;
7) Drag coefficients for balloon, aeroshell, and parachute are assumed as constant values. Mach number and Reynolds number effects are considered of second order importance;
8) The sun angle is assumed constant during the time span of the problem;
9) No atmospheric heating is assumed to occur during balloon inflation. This is a conservative assumption, implying that the balloon gas will heat up somewhat faster than indicated in this program.

The program generates balloon volume, weight of inflation gas, and weight of inflation tanks that will float a specified gondola weight at a specified density. Atmospheric pressure and temperature at the float altitude as well as balloon skin density per unit volume are input.

In the process of determining the dynamics of the system, the temperature of the gas in the balloon, the balloon skin temperature (4 nodes), film temperatures inside and outside the balloon skin, and the inflation tank temperature are computed.

The balloon may float in and out of clouds by input of a cloud top level that varies with time. The state of being in or out of the clouds determines the proper heating equations to be used. Horizontal wind is input as a function of altitude, whereas vertical wind is input as a function of time.

Requirements and Criteria

The design requirements for the baseline flotation system, based on the three missions under consideration, are as follows:

1) The balloon shall be a spherical, superpressure design of a size to support a 175-lb gondola at a density of 0.06588 lb/ft^3. This density corresponds to a radius of 6108 km in the nominal Venus atmosphere model. (The model atmosphere is described in Appendix A of this report.) The environment at this altitude is summarized in table 74.

TABLE 74.- ATMOSPHERIC VARIATIONS

Atmosphere, % CO_2	Radius, km	Pressure, mb	Mass density, lbm/ft^3	Temperature °F	Temperature °K
85	6107.5	610.9	0.06588	58	288
90	6108.0	612.4	0.06588	70	294
95	6108.4	616.7	0.06588	83	302

2) The balloon shall be designed for a maximum superpressure of 10 mb. This shall produce a maximum film stress of 6000 psi. The nominal superpressure will be 6 $\pm$ 2 mb, which results in an operating film stress of 3600 psi.

3) The balloon shall be capable of being deployed and inflated at an external dynamic pressure of 1 psf at a radius between 6110 and 6106 km.

4) The balloon shall be capable of being inflated with gas within 45 sec.

5) The balloon shall allow a single point attachment of a parachute at its apex and shall facilitate release of this parachute by pyrotechnic or mechanical means.

6) The balloon diffuser sock shall be capable of supporting a shock load of 500 lb resulting from a free fall of 30 ft under 1.0 g.

7) The balloon shall be capable of being packed with a packing factor of 5.0 (ratio of packed volume to that of solid material density).

8) The balloon shall be a superpressure device with a pressure relief system to avoid overstressing the balloon skin resulting from excursions in altitude due to updrafts, and balloon gas temperature increase.

9) The balloon shall be designed to survive and operate as a superpressure device for 100 hr minimum.

10) Materials should have a minimum change in physical characteristics such as strength, fold, tear, optical and thermal properties after handling, sterilization, compacting, and enduring long-duration storage, and the entry environment.

11) The balloon material and finish shall not allow more than 10^{-6} scc/sec/ft^2 of hydrogen permeation at 6 mb pressure differential.

12) The balloon shall contain provisions for mounting pressure and temperature sensors in locations conducive to obtaining accurate measurements during inflation and flotation.

13) The balloon diffuser sock shall allow for attachment of two resistance thermometers approximately 1 ft from the inflation fitting.

Configuration

The general flotation system is depicted in the schematic of figure 76. The system basically consists of the balloon sphere assembly, the packaging canister with superpressure controls attached, and the inflation subsystem.

The balloon sphere is inflated with cold hydrogen gas that warms and is controlled at a superpressure of 6 mb, and the sphere is sized to support the gondola, the balloon itself, and the inflation gas. The resulting 18-ft sphere supports a total weight of about 200 lb. The load skirt distributes the gondola weight (175 lb) to the balloon over a large area. The inflation diffuser sock performs two functions -- first, the sock transmits the parachute loads directly to the canister and gondola when the balloon is initially deployed so that high loads are not experienced by the balloon material; and second, the sock diffuses and distributes the high-pressure inflation gas during the rapid inflation process. The sock consists of a porous dacron sleeve capable of carrying the necessary load, plus a nonporous liner extending about 2/3 of the length from the bottom. The liner assures that the balloon will initially inflate with a bubble at the top, thus resulting in a stable external aerodynamic shape that will minimize balloon material flapping while partially inflated.

The basic balloon has a reinforced cap at the top that has fittings attached to transfer the parachute loads directly to the inflation diffuser sock that in turn is attached to the canister and gondola.

The canister stores the balloon before deployment, retains the deployed diffuser sock and balloon attachments, supports the diffuser nozzle, and supports the balloon pressure transducer and pressure relief hardware.

The inflation system consists of four high-pressure storage tanks manifolded through appropriate tubing, valves and controls to the gas inflation diffuser nozzle.

As the BVS descends on its parachute, a barometric pressure switch activates the deployment and inflation system at the preset pressure altitude. The balloon is released from the canister, deployed by the parachute to its full length, and the inflation gas is released into the balloon. After the balloon is inflated, the tanks, associated tubing, and valves are separated by ordnance-operated cutters and dropped free of the gondola.

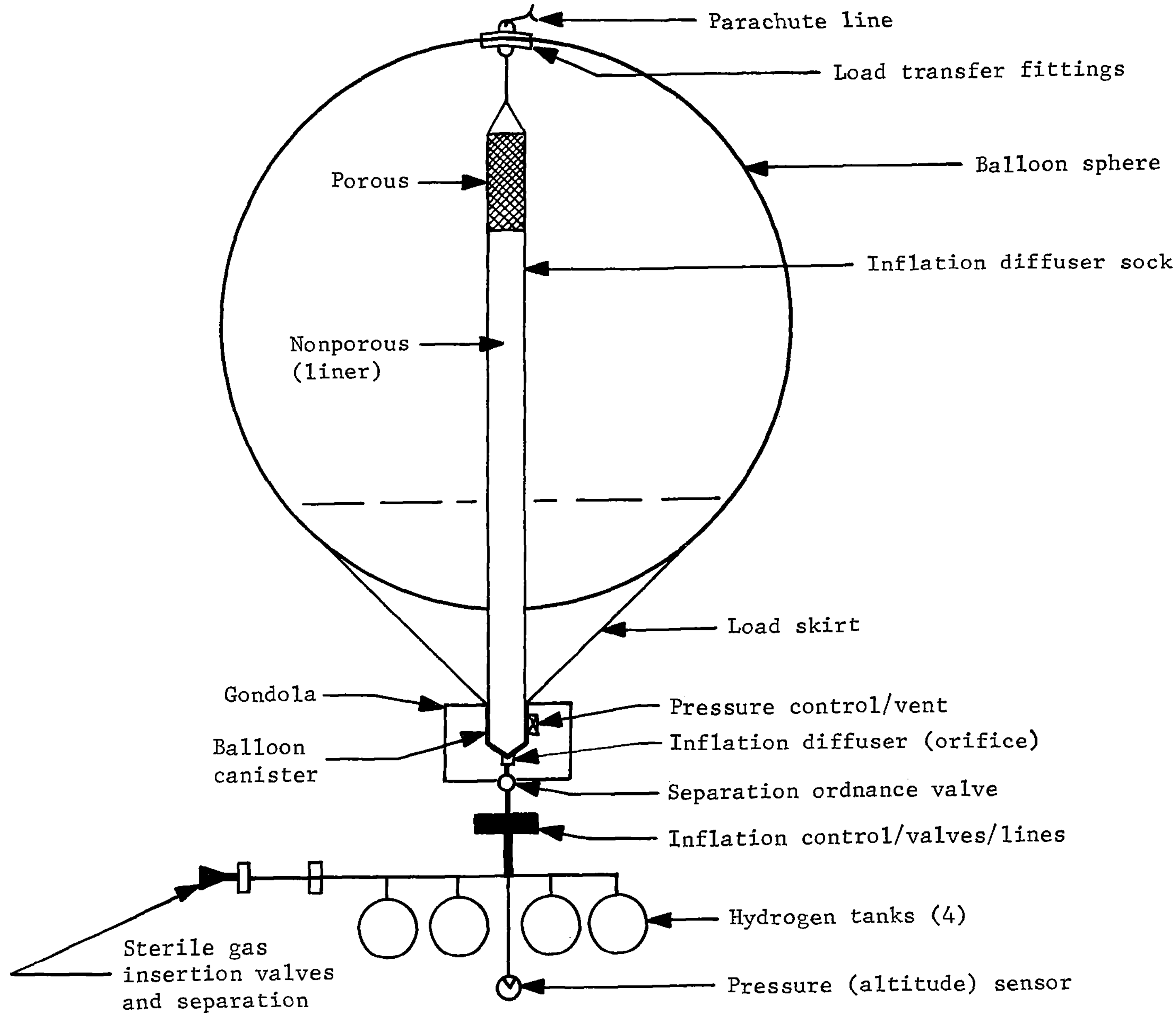


Figure 76.- Flotation System

The gondola supports the mission equipment in an annular structure mounted around the balloon canister.

A summary of pertinent data for the balloon and inflation system is presented in table 75.

TABLE 75.- BALLOON AND INFLATION SYSTEM SUMMARY

Balloon diameter (inflated), ft	17.98	
Balloon volume, ft^3	3 043.74	
Minimum skin thickness required, mils	1.3	
Total laminated skin thickness, mils	1.75	
Balloon film density, lb/ft^3 [a]	98.3	
Balloon sphere weight, lb		14.63
Load skirt weight, lb		.75
Diffuser sock assembly weight, lb		.85
Load fittings weight, lb		.15
Total inflation gas weight, lb	10.06	
Gas in balloon weight (at equilibrium), lb		9.14
Type of gas	Hydrogen	
Weight of gondola, lb		175.00
Weight lofted, lb		200.52
Vented gas, lb		.92
Volume of four gas tanks, $in.^3$	12 633.3	
Weight of four gas tanks, lb		128.0
Weight inflation plumbing and controls, lb		16.0

[a] Increased 13% to account for weight of construction tapes, seams, etc.

Balloon Design

The balloon design* shown in figures 77 and 78 is fabricated of 18 gores of 3/4x3/4-mil Mylar laminated with 1/2 mil of adhesive. The balloon skin tailored to meet the design stress is a 0.625x0.625-mil Mylar laminated with 1/2 mil of adhesive. The gore seals are formed with GT 300 tape on both interior and exterior surfaces. Inflated to 6 mb superpressure, the diameter is 18.05 ft. The 0.77 ft^3 allowed for the stored balloon -- before deployment -- results in a packing density of 23.92 lb/ft^3, which can be achieved without auxiliary mechanical pressure.

The sphere will have an end cap in the apex and base. The end caps are required for load transition across the gore end points, for load transfer fitting mounting base, and for mounting the base duct tube. The end caps are sealed into the sphere with GT 100 tape. A layer of GT 300 tape is applied to the exterior transition point between the sphere gores and the end cap. The apex and cap are 12 in. in diameter and require an 8-in. diameter port in the balloon apex for mounting. The base end cap is 24 in. in diameter and requires a 20-in. diameter port in the balloon base.

The base end cap and tube mounting requires special fabrication procedures. The tube is inserted into the end cap so that a 3 in. length of tube protrudes into the balloon interior. Sealing is accomplished by overlapping 6-in. lengths of GT 300 tape. The tape is placed with the 6-in. length parallel to the axis of the tube, 3 in. being sealed to the tube, and 3 in. being sealed to the end cap. The same technique is used on the exterior surfaces.

The apex end cap is installed with the load transfer fitting in place.

The load transfer fitting is fabricated from a molded, glass-filled nylon. This material was selected for resistance to ethylene oxide and for absence of rf attenuation characteristics. The fitting is threaded to facilitate installation. Sealing around the access port will be accomplished with an integral Buna-N gasket. The load transfer line is threaded through the fitting loops and spliced.

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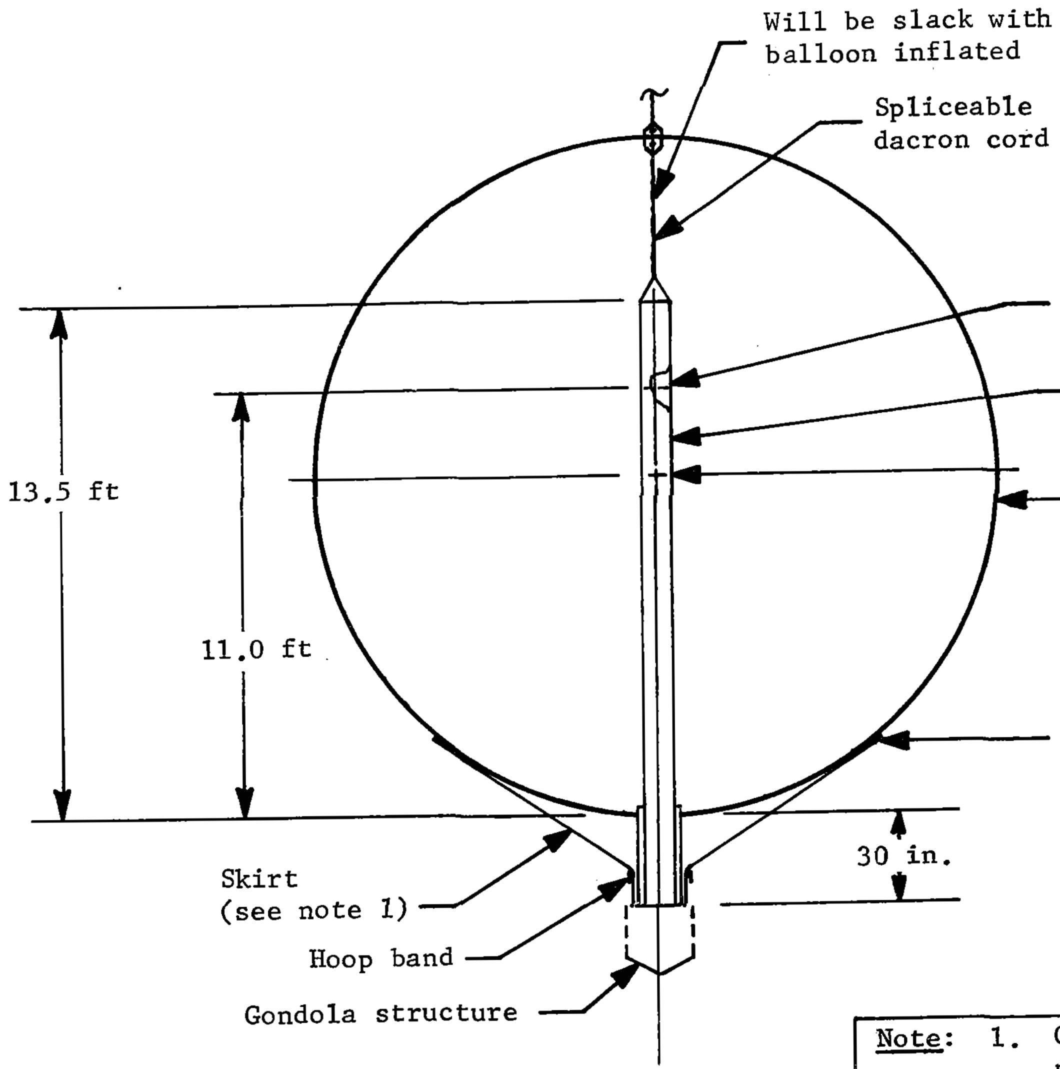
*Raven Industries, Inc., J. C. Poland, C. A. Schanche, and K. D. Odney.



[a] Raven Industries, Inc., Drawing Number 07532.

Note:
1. Cloth, dacron, ripstop, natural, J. P. Stevens & Co. S/N-24681/1, 1.25 oz/yd^2.
2. Mylar 3/4 mil x 1/2 mil adhesive x 3/4 mil trilam.

Figure 77.- BVS Balloon Assembly[a]

10	A/R		GT 300 1 x 1 x 1-1/2 x 6" long tape
9	1		Attachment lug assy
8	1		Tube 13" od, 3/4 mil x 1/2 mil adhesive x 3/4 mil trilam Mylar
7	A/R		GT 100 2-1/2 x 2" tape
6	A/R		GT 300 1/2 x 1/2 x 3/4" tape
5	A/R		GT 300 1 x 1 x 1" tape
4	A/R		GT 300 1 x 1 x 1-1/2" tape
3	1		End cap - base, 1 mil x 1 mil bilam Mylar
2	1		End cap - top, 3/4 mil x 1/2 mil adhesive x 3/4 mil trilam Mylar
1	18		Gore, 3/4 mil x 1/2 mil adhesive x 3/4 mil trilam Mylar
Item no.	No. reqd	Part no.	Description

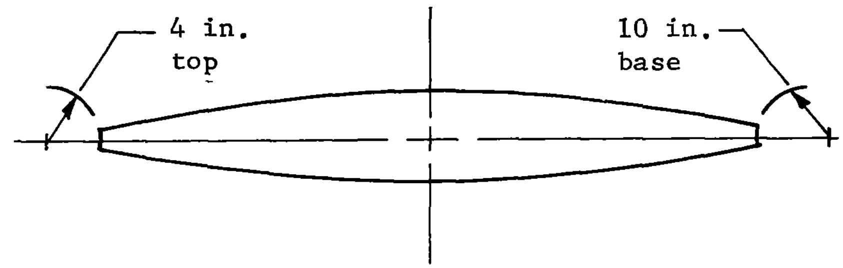
(a) Sheet 1

Note: 1. Diameter of sphere: 18.05 ft (inflated to 10 mbΔP).
2. Number of gores in sphere: 18.
3. L = one quarter of the cir of sphere = one gore length;
S = gore length segments for plotting pattern;
W = width of gores;
1/2W = one-half width of gores at "S" points.

[a] Raven Industries, Inc., Drawing Number 07530.

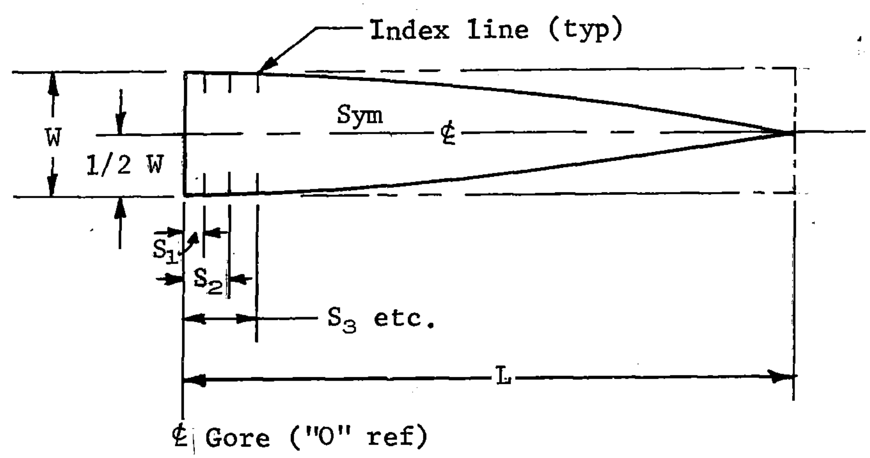
Figure 78.- 18.05-ft Diameter Sphere Assembly[a]

Sta. no.	S, ft	½ W, ft
0	0	1.566
1	.944	1.557
2	1.889	1.532
3	2.834	1.489
4	3.778	1.431
5	4.723	1.356
6	5.667	1.267
7	6.612	1.164
8	7.557	1.048
9	8.501	.920
10	9.446	.783
11	10.390	.637
12	11.334	.484
13	12.280	.326
14	13.224	.164
15	14.169	0



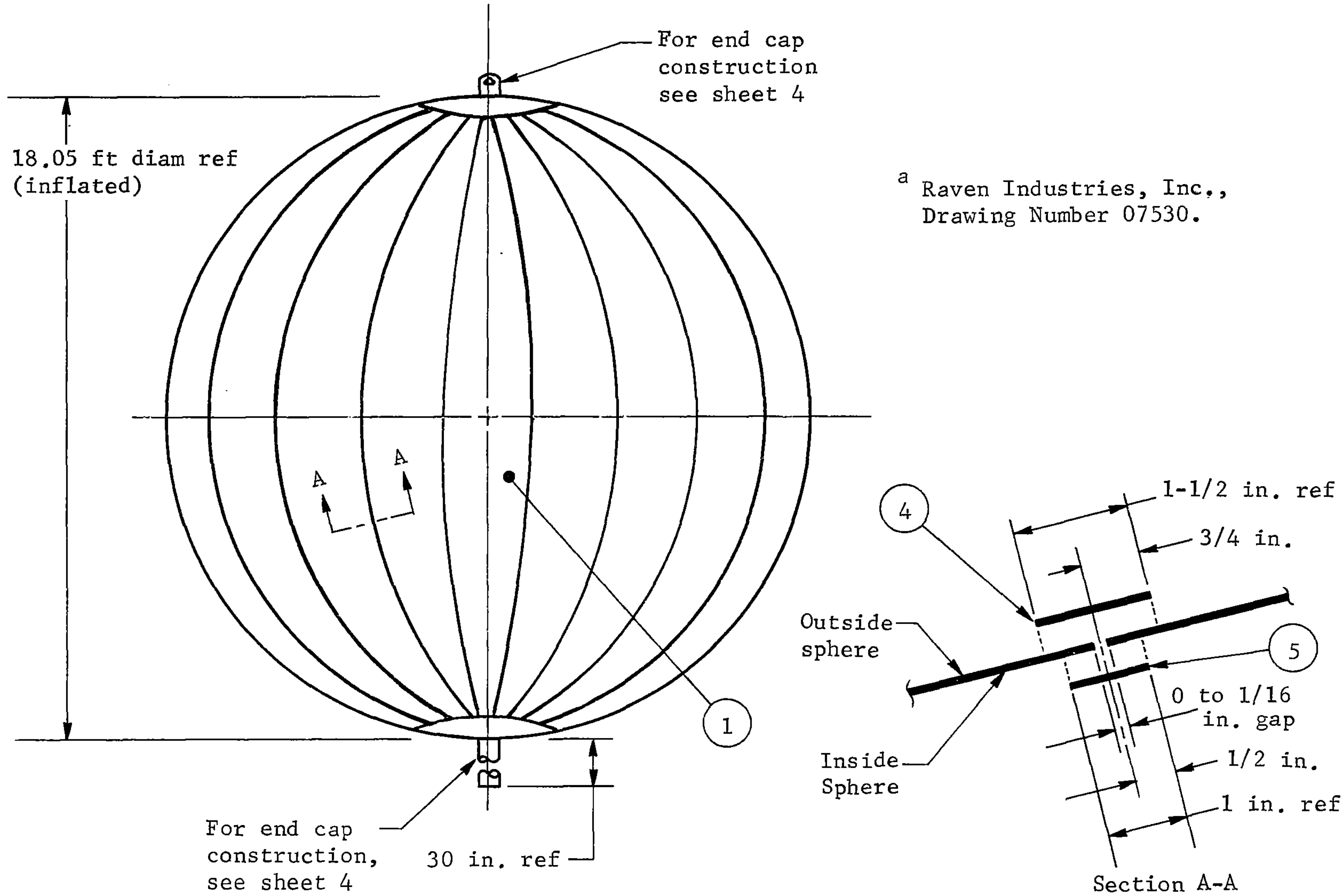
Sphere gore template
(17.95 ft diam)

[a] Raven Industries, Inc., Drawing Number 07530.



(b) Sheet 2

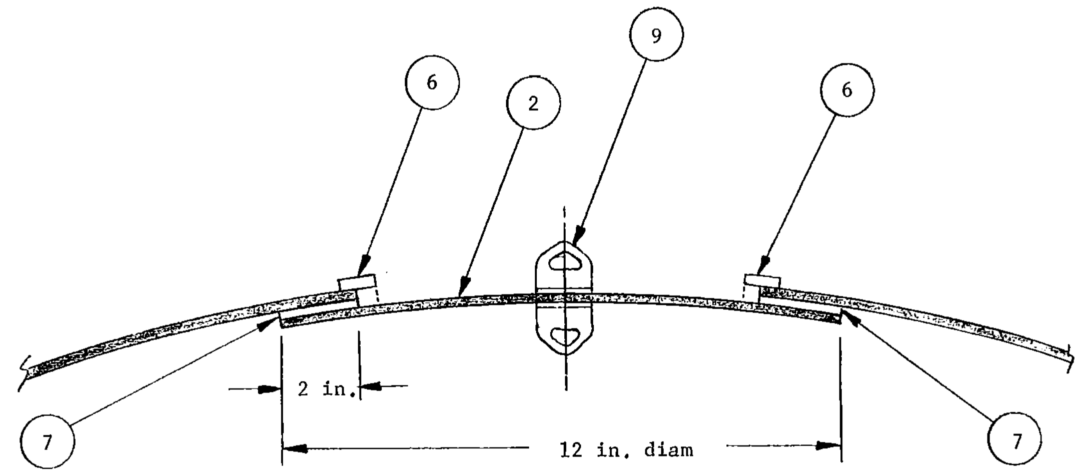
Figure 78.- Continued[a]



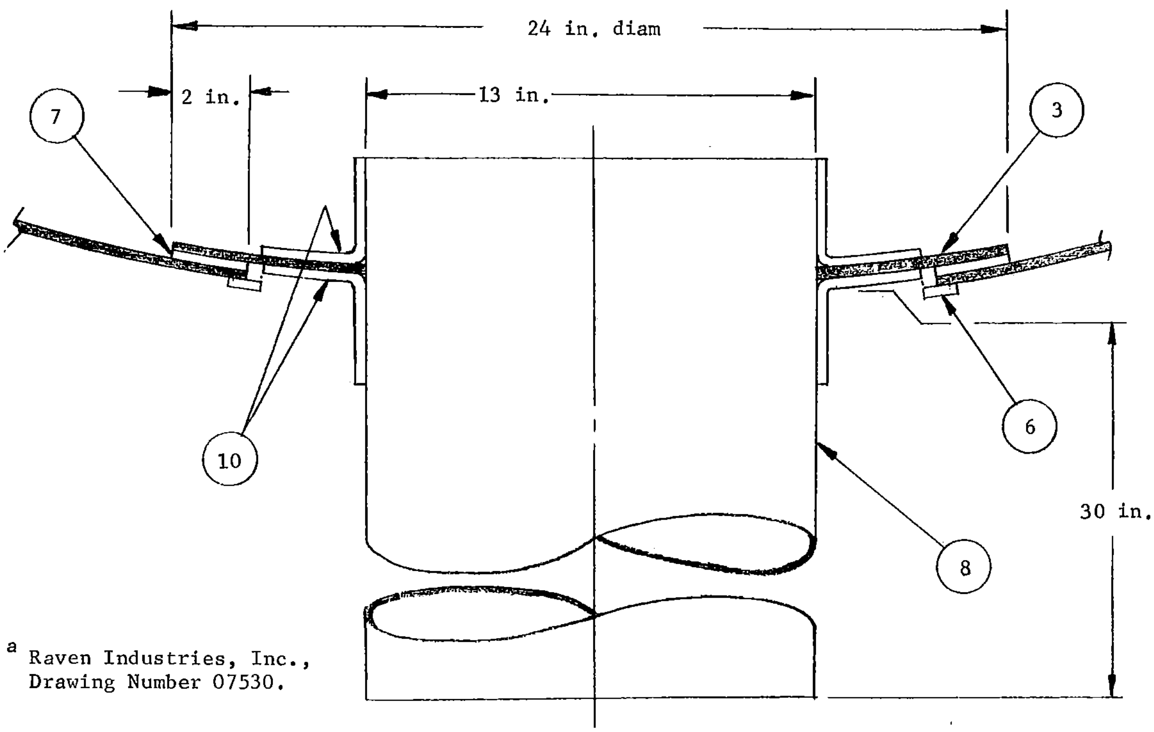
[a] Raven Industries, Inc., Drawing Number 07530.

(c) Sheet 3

Figure 78.- Continued[a]



Top end cap construction



[a] Raven Industries, Inc., Drawing Number 07530.

Bottom end cap construction

(d) Sheet 4

Figure 78.- Concluded[a]

The load support skirt is fabricated from 1.25 oz/sq yd dacron gores sewed together. The skirt is bonded to the balloon skin with an adhesive similar to Bostik 7064. This adhesive is used for bonding load patches on present superpressure balloons. However, the resistance of this adhesive to the sterilization environment is not known.

Figure 79 shows the configuration of the lower balloon area and balloon can. The inflation diffuser sock is the load-carrying member from the gondola through the balloon to the parachute before balloon inflation. It also serves to diffuse the gas flow into the balloon. The diffuser sock is designed as a sealed tube from the balloon can to a point near the top of the balloon, forcing the gas to the top of the balloon, filling the top first. After balloon inflation, the load is carried into the balloon via the support cone, relieving the inflation sock of any load.

Figure 80 shows the top of the balloon and how it is attached to the parachute system. The pyrotechnic cutter separates the parachute from the balloon just before the completion of the balloon fill cycle.

Balloon Materials

The balloon system material that appears the most promising at present is Mylar. Until the much-needed materials sterilization tests and other pertinent environmental tests can be conducted, Mylar will be considered the baseline material. Mylar's properties as affected by temperature for various time durations have been fairly well established by the E.I. duPont DeNemours & Company and are discussed later. However, the effects of decontamination by use of ethylene oxide before dry-heat sterilization have not been determined. Other environmental factors may be even more significant, and testing will be required to evaluate these effects and various combinations of effects. Following is a list of typical environmental conditions the balloon material may experience:

1) Manufacture of material;
2) Construct balloon;
3) Fold and ship;
4) Decontaminate;
5) Evacuate balloon, fold, compress, and pack in balloon canister;

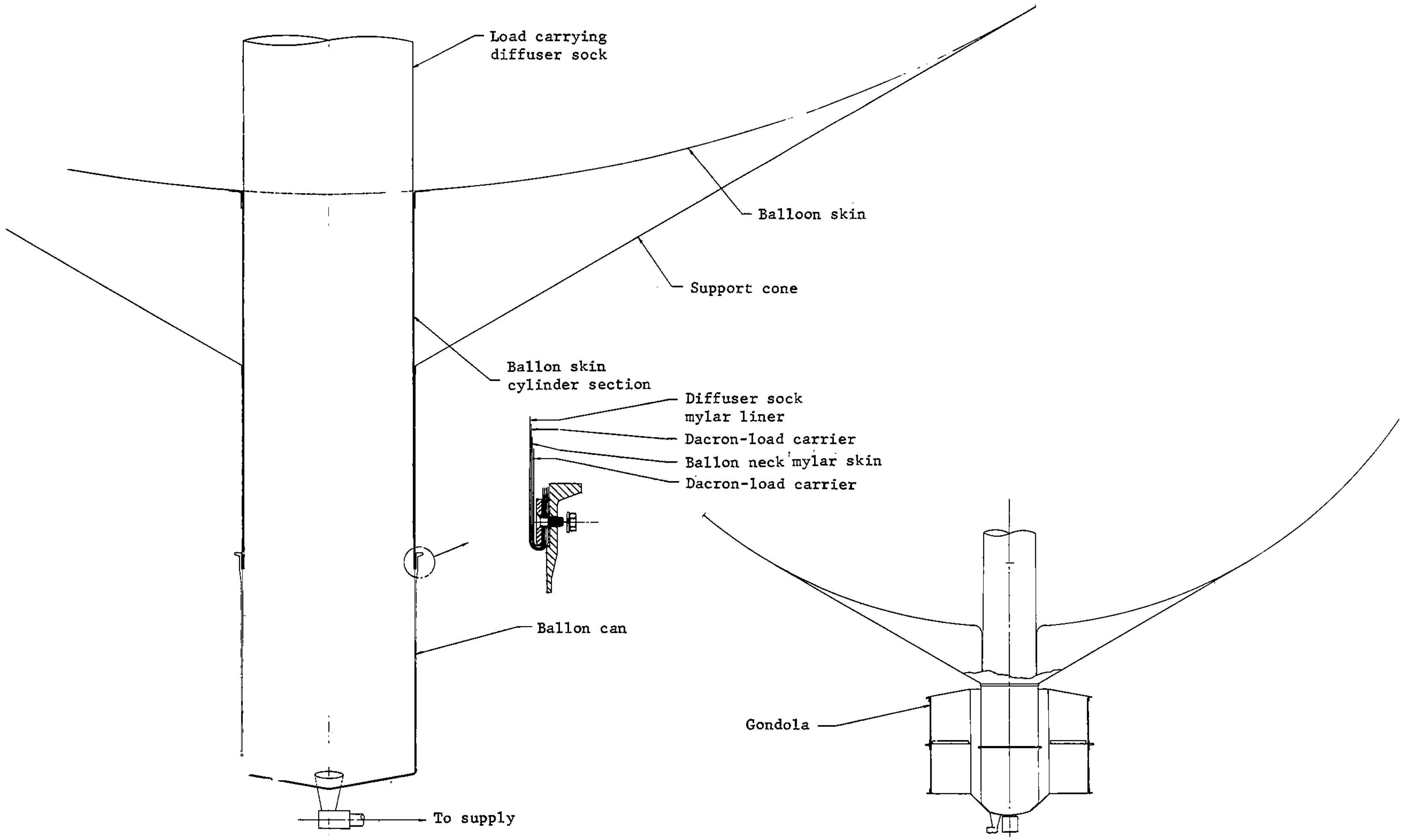


Figure 79.- Ballon Base Gas Diffuser Layout

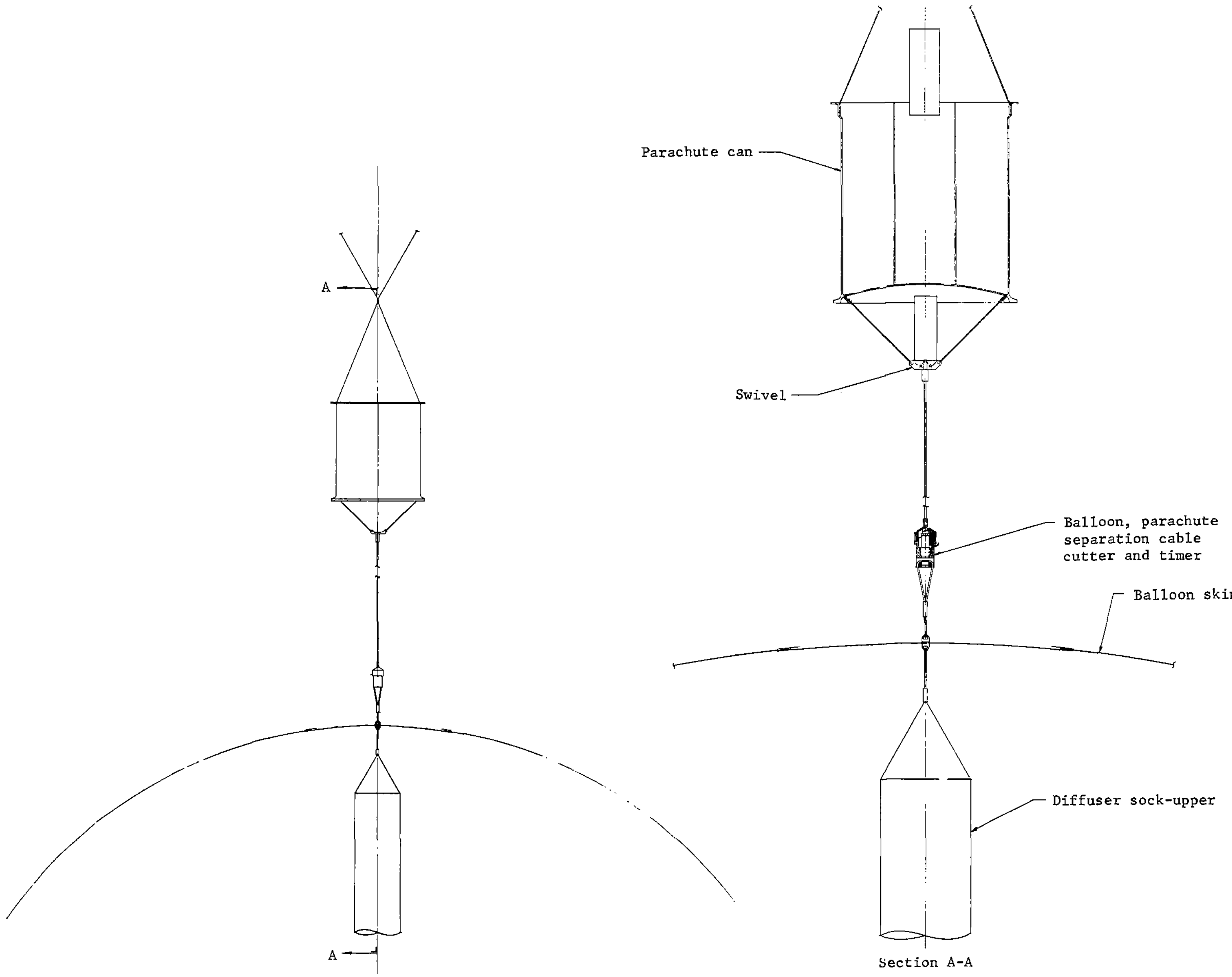


Figure 80.- Ballon to Parachute Can Configuration

6) Dry-heat sterilize;

7) Launch to Venus;

8) Effects of canister pressure, temperature, and storage gas for mission duration;

9) Endure 170 to 340 g entry deceleration at nominal temperature in balloon canister;

10) Extract balloon by parachute with loads going into balloon inflation sock;

11) Endure dynamic pressure (1 psf) during inflation;

12) Endure inflation gas temperature of $-60^{\circ}F$ minimum.

Appropriate tests must eventually be conducted to simulate the environment history to fully evaluate a material. In addition to the physical properties such as strength and pin hole/leak characteristics, it will be necessary to establish the optical/thermal properties such as absorptivity, emissivity, transmission, and radiation properties that directly affect the balloon gas equilibrium and transient temperatures as shown later in this report.

Tests are being conducted at Martin Marietta Corporation on a variety of materials and laminate combinations. Kapton is also promising and is slightly more heat resistant than Mylar; however, within the present state of materials testing, Mylar remains the preferred material.

To overcome the problems of material damage due to folding and compacting, an alternative approach looks very promising. A laminate consisting of an outside layer of fine dacron fabric for strength and flexibility with either a thin Mylar -- flexible layer-Mylar laminate inside -- or possibly only a flexible layer-Mylar inner laminate would provide a gas barrier (not subject to pinholes), and the inner Mylar would provide scuff protection to the barrier material.

Balloon Storage Canister

The balloon canister is shown in figure 81. The concept involves maintaining pressure in the canister at approximately the ambient pressure at time of balloon deployment ($\sim$9 psia) to eliminate vacuum effects on the balloon, i.e., outgassing could cause destructive bubbles in the packed balloon.

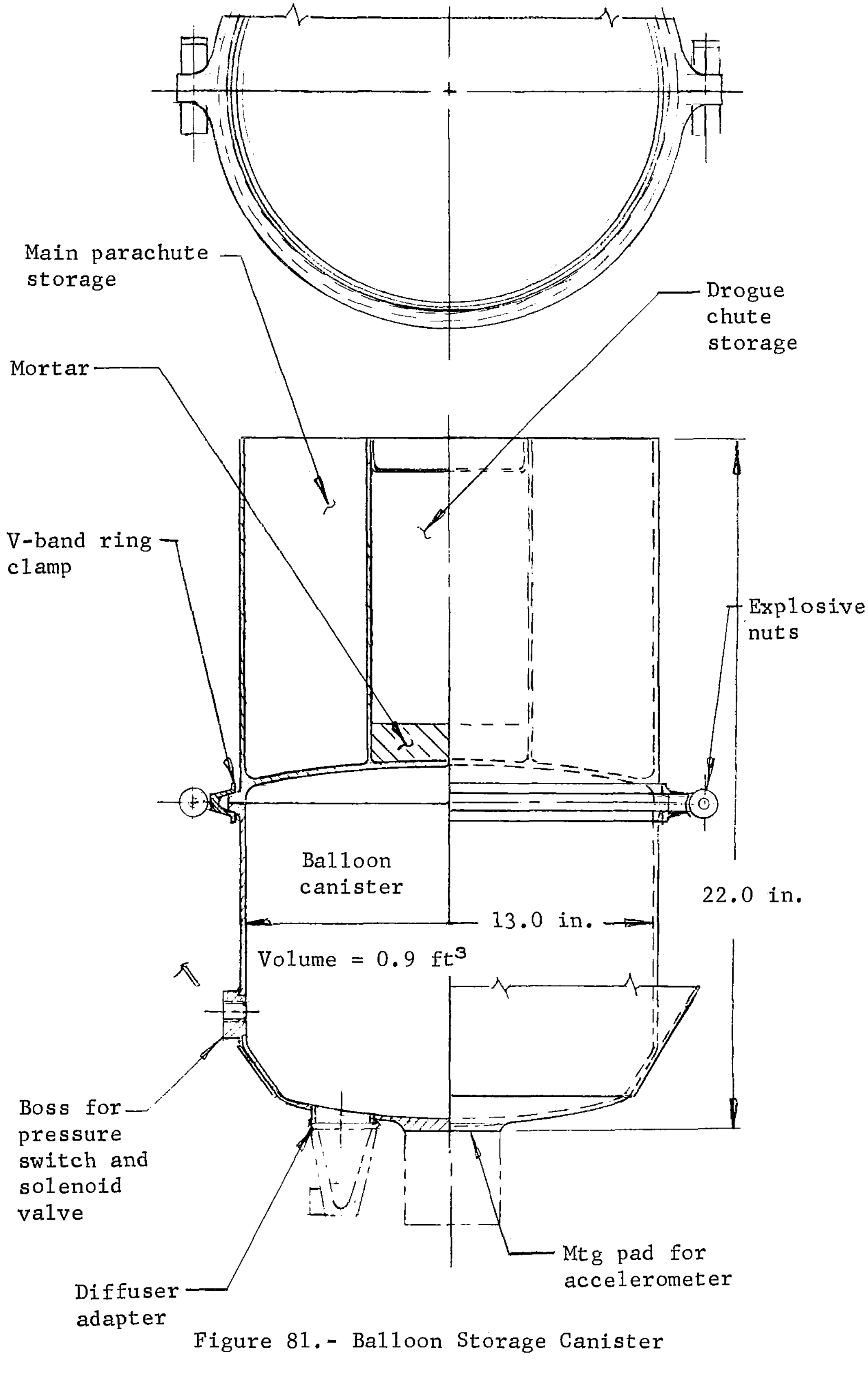


Figure 81.- Balloon Storage Canister

The canister is cylindrical in shape. The lid is the base of the parachute can and contains a harness and swivel to relieve possible twist loads developed by the parachute.

The pressure sensor and pressure relief valve are located on the side of the canister and the inflation diffuser is located on the base. A V-clamp or Conoseal band clamp retains the lid and is released by a pyrotechnic device. The seal is either an aluminum band or an O-ring.

A series of accordian type folds is used in packing the balloon and sock into the storage canister. The objective is to eliminate any twist loads or material hangup within the balloon folds themselves. The balloon extraction is rapid; however, the actual parachute snatch load is taken up by the internal diffuser sock. As long as the material folds do not stick, the balloon material should not receive any high loads.

Balloon Controls and Instrumentation

The basic balloon flotation control consists of a sensitive pressure switch (6 $\pm$ 2 mb) that actuates the balloon vent (solenoid) valve. Both the switch and vent valve are mounted on the canister that initially contains the packed balloon and diffuser. The vent valve remains shut during the balloon inflation process by being electrically locked out during inflation, because the local pressure in the canister will be considerably higher than 6 mb at that time. When the hydrogen is completely expelled into the balloon, the vent valve and pressure switch circuit are activated and maintain the proper superpressure during the mission.

An example of the type of pressure switch required for this application is represented by the Consolidated Controls Corporation low-level pressure switch shown in figures 82 and 83. This switch is capable of doing the balloon pressure control job as specified; however, the switch is not qualified for sterilization and other specific environmental conditions. Therefore, the flight switch would require a qualification program with some possible modifications.

The 1 in. low-pressure drop, solenoid-actuated vent valve, although not qualified, is available with minor modifications from Valcor Engineering Corp.

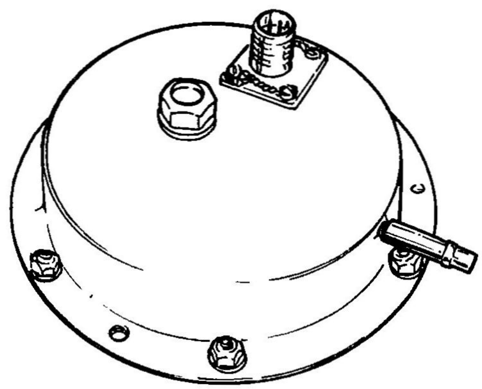


Figure 82.- Low-Level Pressure Switch

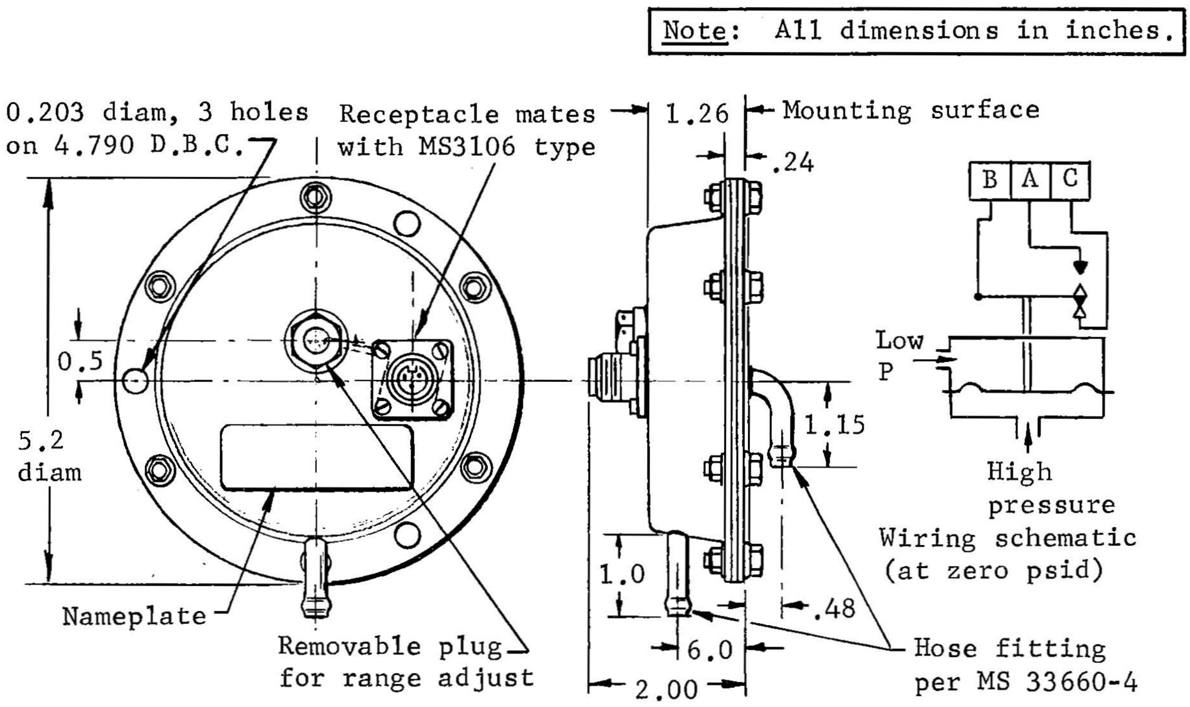


Figure 83.- Pressure Switch General Specifications

The balloon instrumentation consists of pressure and temperature sensors mounted in locations conducive to obtaining accurate measurements during inflation and flotation. Two resistance thermometers are attached to the inflation diffuser sock about 1 ft from the inflation fitting.

BVS requires special control hardware to accommodate the loss in the balloon supertemperature if the BVS crosses the terminator to the dark side of Venus. It is assumed that the balloon remains in the clouds as it crosses the terminator and when the solar radiation drops to zero the balloon cools down to ambient temperature. The ambient temperature, pressure, and density are assumed to remain the same on either side of the terminator within the clouds.

This cooling results in a loss of essentially all of the supertemperature, and the balloon will partially collapse. The volume reduction results in negative free lift and the balloon will descend to the surface unless corrective action is taken.

As discussed later in this report, corrective action can be in the form of either dropping the two small sondes as ballast or by using a gas makeup system with about 0.5 lb of gas. For the ballast or drop sonde release, a control system is required that will delay corrective action so that a premature weight drop will not occur resulting in further gas venting. Makeup gas can be slowly injected by using a simple blowdown orifice-controlled system.

Various makeup systems are compared, but the high-pressure, gaseous hydrogen in a storage tank appears to be the best approach.

Inflation Subsystem

The requirements to be satisfied by the inflation system are summarized in table 76.

TABLE 76.- SUMMARY OF INFLATION SYSTEM REQUIREMENTS

1) Provide storage for 10.2 lb of hydrogen inflation gas;
2) Provide valving and flow controls to,
a) Complete pressurization of the balloon within 45 sec,
b) Limit the balloon pressure to 6 mb above ambient pressure;
3) Provide capability for separation of the inflation system from the gondola;
4) Provide capability for sterilization in the unpressurized condition and for sterile loading of the hydrogen;
5) Provide instrumentation to measure gas temperature, pressure, and flow rate during inflation;
6) Provide filtration to protect the balloon from particles greater than 10 μ in size.

The baseline inflation system designed to satisfy these requirements is shown schematically in figure 84. This system is described in the following paragraphs. Refer to the structures section of this report for the detail layout of this subsystem.

The gas storage system, consisting of four manifolded storage tanks, is loaded after sterilization. Loading is accomplished through a 1/4-in. line that passes through the biocanister and aeroshell. A disconnect is located on the external surface of the biocanister, and the fill line penetrates the biocanister and the aeroshell. Tube cutters are located on both sides of the aeroshell to enable the fill line to separate from the aeroshell and biocanister. Normally open ordnance valves inside the aeroshell provide a zero leak seal for the inflation system. The system is pressurized slowly to maintain the storage tanks at less than 150°F as monitored on the storage tank temperature transducer. After temperature stabilization, a valve is closed in the external system and the pressure is monitored until essentially zero leakage is verified. Because numerous detailed leak checks of the system joints with a helium mass spectrometer are accomplished during fabrication and before sterilization, a high degree of confidence that the joints are leaktight is established before this final check.

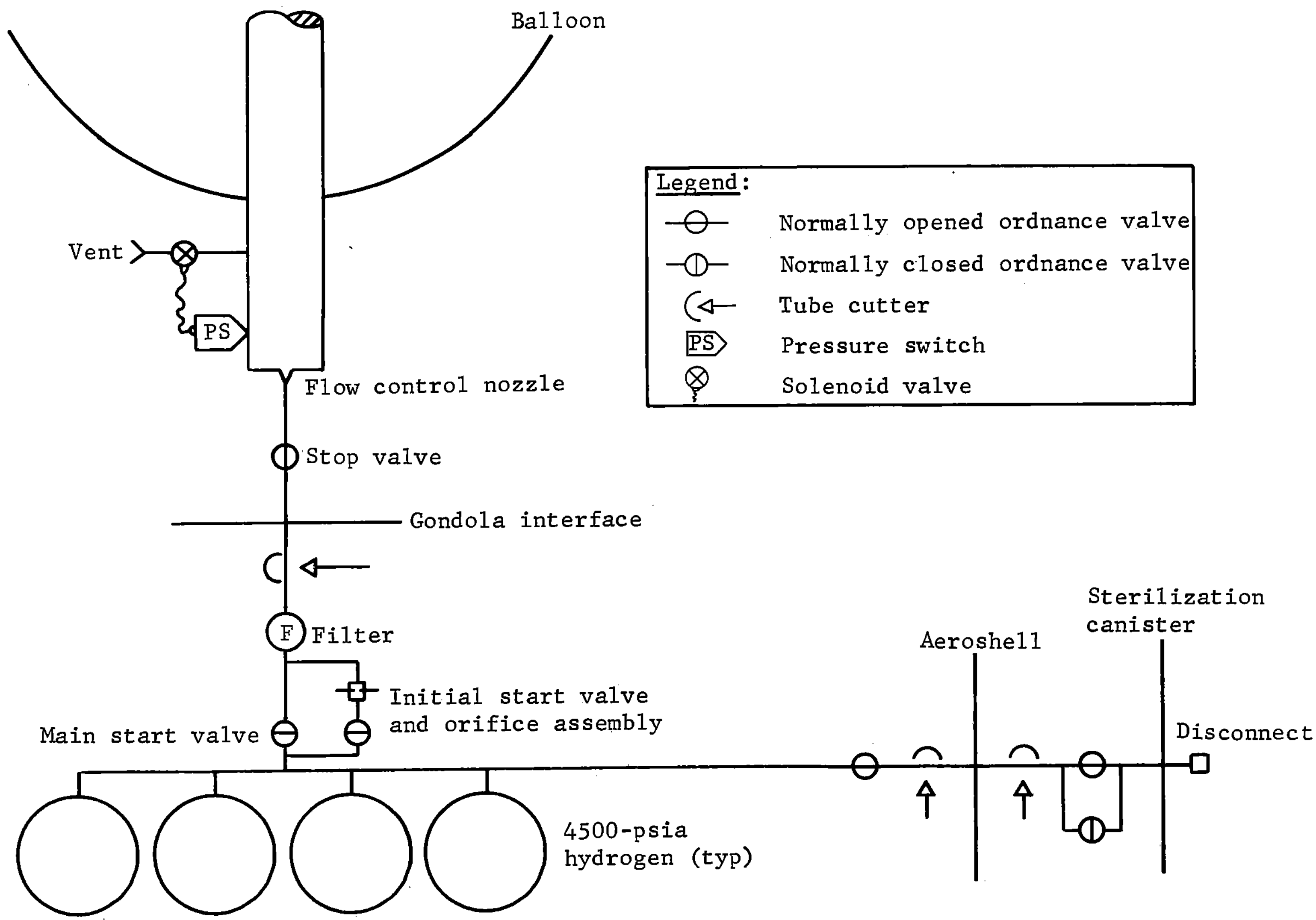


Figure 84.- Inflation System

Before launch countdown, the normally open valve mounted on the inside of the biocanister is closed, and the external line is disconnected and capped. In case of an abort at any time before the final automatic launch sequence, the system can be depressurized by reconnecting the external lines and opening of the normally closed unloading valve. During the final launch sequence, the normally open valve inside the aeroshell is closed, followed by operation of the two line cutters. Current traces are used to establish proper function of the ordnance devices. At this time, the system is sealed and separated from the aeroshell and the biocanister and is considered ready for flight and balloon deployment.

The inflation sequence is initiated by a pressure switch sensing atmospheric pressure. At an ambient pressure of 612 mb, the switch contacts close causing a signal to be sent to the sequencer. The sequencer, in turn, sends a signal to the low-flow start valve initiating the flow of hydrogen gas to the balloon. The low-flow start valve is a normally closed, ordnance-operated valve that supplies hydrogen gas from the 4500 psig storage system through a metering device to the balloon inflation system. The hydrogen gas is introduced into the balloon through a nozzle attached to the bottom of the balloon canister. After 2 sec of this low hydrogen flow, the balloon gas diffuser is partially inflated, and the sequencer signals the high-flow start valve open. This valve is mounted in parallel with the low-flow valve and is also a normally closed ordnance-operated device. Upon opening, the high-flow rate blowdown of the hydrogen gas storage tank is initiated. At this time, the nozzle at the balloon canister flows sonically, discharging hydrogen gas into the sleeve at approximately 2.0 lb/sec. This blowdown of the hydrogen storage continues for a total duration of 45 sec, at which time the correct amount of hydrogen has been introduced into the balloon. A sequencer signal causes the system shutoff valve to close. It should be noted that a line filter is located between the ordnance-operated start valves and the gas nozzle to prevent introduction of particles generated by the ordnance start valves into the balloon.

After closure of the shutoff valve (1 sec), another signal is sent by the sequencer to effect line disconnect at the pressurization system/gondola interface. This action is accomplished by an ordnance-operated tube cutter that severs and releases the hydrogen inflation line at a point that allows complete separation of the inflation system and the gondola without mechanical interference. The release of the pressurization system occurs 1 sec after line disconnect again by a signal from the sequencer to the ordnance-operated explosive bolts (2) at the gondola/inflation system interface.

At the completion of the inflation sequence, the balloon overpressure relief system is armed. This system consists of a 6-mb gage pressure switch sensing balloon pressure with respect to atmospheric pressure. It is connected to a solenoid-operated relief valve having an equivalent orifice size of 1 in. An increase in balloon overpressure above 6 mb results in opening of the valve with subsequent venting to maintain the 6-mb superpressure.

The components that comprise this system and their characteristics are shown in table 77.

The performance of the inflation system is shown in figures 85 and 86. Figure 85 presents pressure and temperature of the gas in the storage tanks vs time. The analysis includes a real-gas equation of state and accounts for the effect of heat transfer from the storage tank to the gas. In figure 86, the weight and temperature of the gas in the balloon is shown as a function of time. It can be seen that the inflation is essentially complete within 30 sec. The temperature of the gas in the balloon is about -45°F at the end of inflation. However, this value is slightly low because the heat transfer from the balloon to the hydrogen was not considered.

Several inflation system options were investigated including hydrogen, helium, and various hot-gas generation schemes. Because the hot-gas system is more complex and, thus, less reliable, it has been eliminated as a candidate for the inflation system. Helium is an alternative because of the advantages of helium in handling and reduced leakage. Both hot-gas generation and helium systems have some weight disadvantages. Results of comparisons are discussed later in this section.

Parachute Subsystem

The design requirements for the parachute subsystem are as described in the following paragraphs.

The parachute subsystem is comprised of two parachutes -- the afterbody parachute is deployed by mortar, the BVS parachute is deployed by the afterbody parachute and separated from it by a break tie.

TABLE 77.- BALLOON INFLATION COMPONENTS

Component	Quantity	Setting/ tolerance	Unit weight, lb	Total weight, lb
Pressure switch (overpressure control)	1	6 $\pm$ 2 mb	0.5	0.5
Pressure switch (inflation initiation)	1	612 $\pm$ 10 mb	0.5	0.5
Solenoid valve (vent)	1	1 in. line size 1 A at 28 Vdc	1.0	1.0
Filter	1	3/8 in. line size 10 μ nominal	0.5	0.5
Ordnance valve (shutoff valve)	1	3/8 in. line size normally open	1.1	1.1
Ordnance valve (depressurization valve and low-flow start valve)	2	1/4 in. line size normally closed	0.4	0.8
Ordnance valve (main start valve)	1	3/8 in. line size normally closed	1.1	1.1
Ordnance valve (fill valves)	2	1/4 in. line size normally open	0.4	0.8
Cable cutter (separation)	1	3/8 in. line size	1.0	1.0
Cable cutter (separation fill line)	2	1/4 in. line size	0.5	1.0
Ordnance squibs	18	----	0.1	1.8
Gas inflation nozzle	1	----	0.5	0.5
Pressure transducer (tank)	1	0 to 5000 psig $\pm$ 2%	0.5	0.5
Pressure transducer (nozzle outlet)	1	0 to 15 psia $\pm$ 2%	0.5	0.5
Temperature transducer (tank)	1	-300 to +300°F $\pm$ 2%	0.5	0.5
Temperature transducer (nozzle inlet)	1	-100 to +150°F $\pm$ 2%	0.5	0.5
Orifice assembly and filter (low flow control)	1	----	0.1	0.1
Storage tanks	4	Volume = 1.9 ft^3	32	128
Hydrogen gas	----	----	10	10
Tubing	----	----	----	5
				154

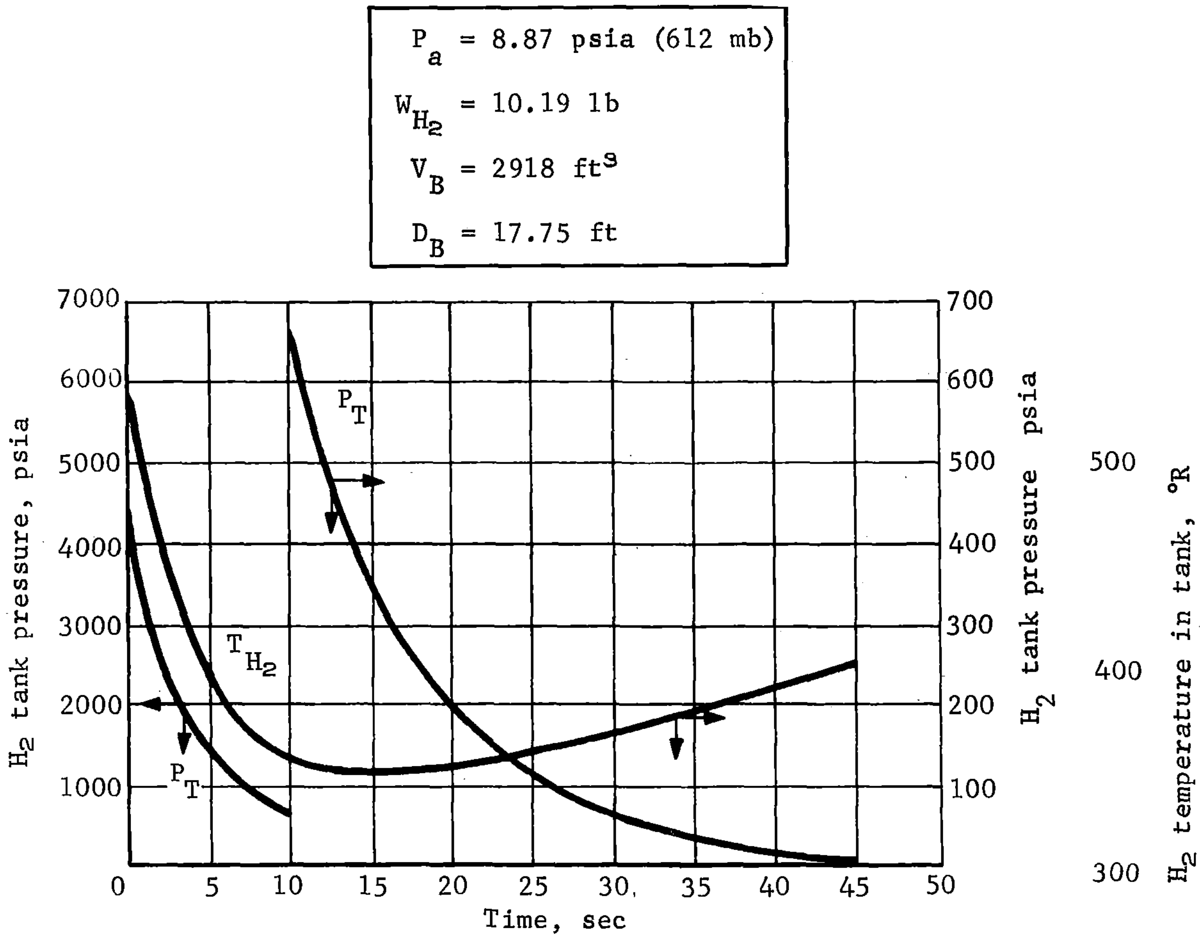


Figure 85.- H_2 Tank Pressure and H_2 Temperature vs Time

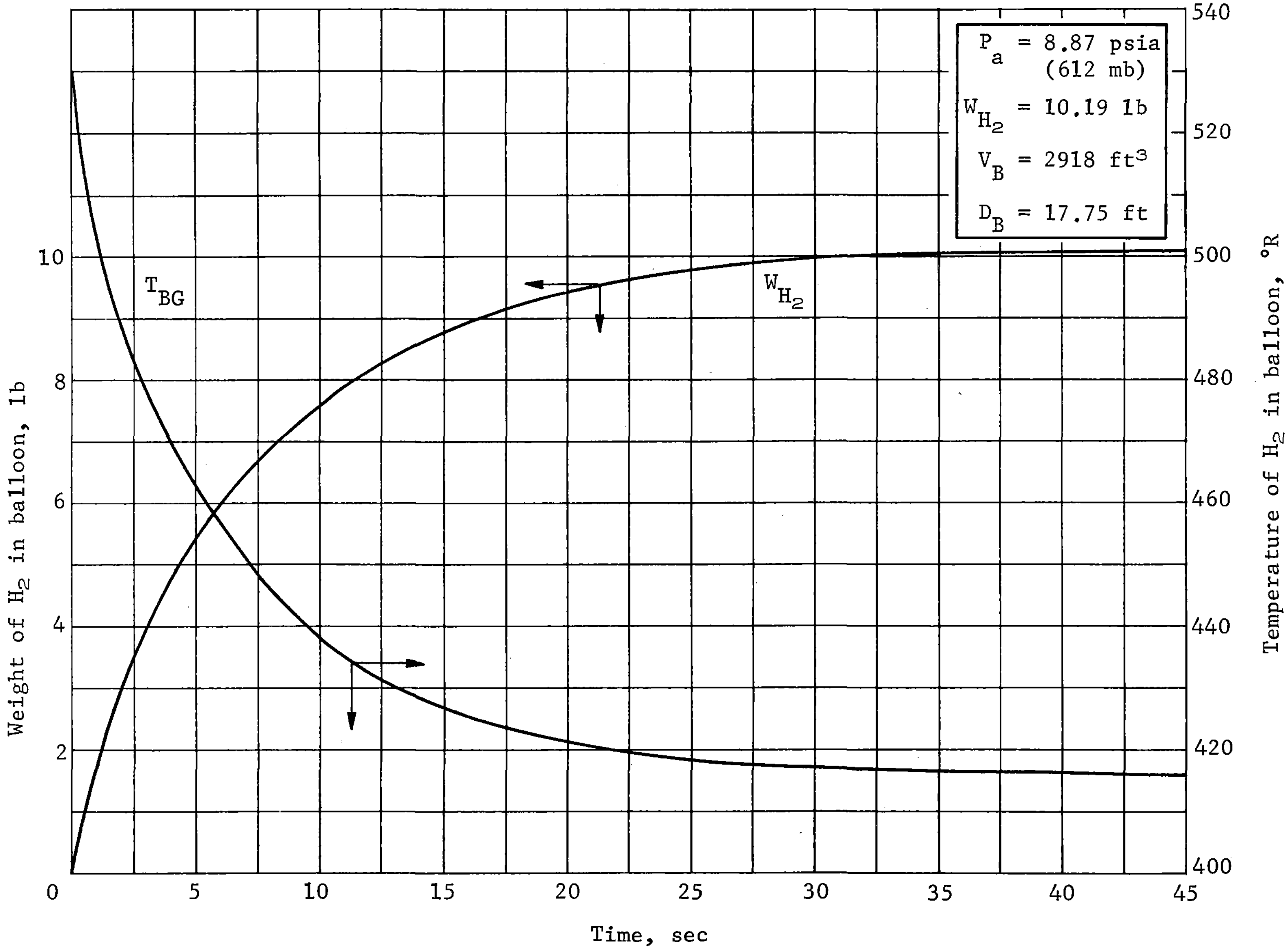


Figure 86.- Weight of H_2 in Balloon and Balloon Gas Temperature vs Time

Afterbody parachute.- A parachute will be provided to extract and retard the aeroshell afterbody. This single stage parachute will be deployed by operation of a mortar, will be capable of opening at $M = 0.50$ with a maximum dynamic pressure of 20 psf, and will be designed for a terminal descent dynamic pressure of 10 psf supporting a load of 100 lb (no drag for load). This parachute will operate properly with an aeroshell spinning at 0.50 rad/sec and will be designed to descend vertically in still air.

BVS parachute.- A parachute will be provided to extract and retard the BVS. This parachute will be capable of opening at $M = 0.50$ with a maximum dynamic pressure of 20 psf, will be designed for a terminal descent dynamic pressure of 1.0 psf supporting a (BVS) weight of 375 lb (no drag for BVS), and will be designed to descend vertically in still air.

The Mission Description Section presents a detailed description of the BVS sequence of events including the parachute and balloon deployment. Briefly, the sequence is as follows: the entry capsule descends to deployment altitude; the afterbody parachute is deployed by mortar and it removes the afterbody which deploys the main chute; and the aeroshell drops away.

The nominal parachute designs resulting from the above design criteria are shown in table 78, and parametric data are shown in figure 87, prepared by the G. T. Schjeldahl Company. The disc-gap-band chute is very stable, but has a weight-to-drag efficiency somewhat lower than a standard flat-type chute. This stability is desirable when releasing the aeroshell and deploying the balloon. The deployment conditions for these parachutes are well within the state of the art, and no unusual problems are anticipated. The packing density of 19 lb/cu ft is easily met, and the parachute container could probably be reduced in size.

TABLE 78.- NOMINAL PARACHUTE DESIGNS

Type	Main chute disc-gap-band	Afterbody chute flat
Diameter, ft	32	10
Weight canopy, lines, lb	12.0	2.0
Opening force, lb	2800	1700
Packing density, lb/ft^3	19.0	19.0
Terminal dynamic pressure, psf	1.0	10

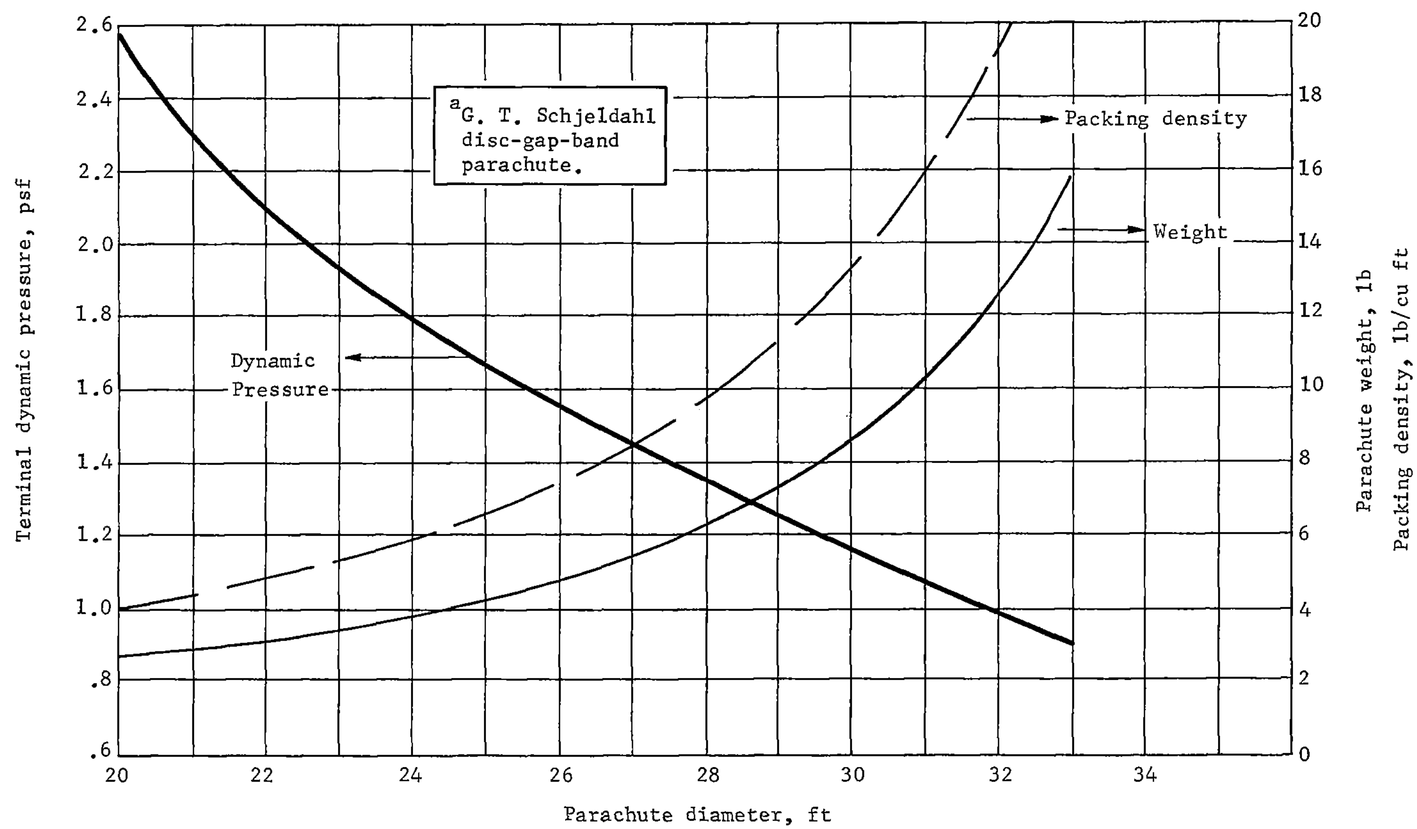


Figure 87.- Parachute Parametric Data[a]

According to the G. T. Schjeldahl Company, the main chute with its canister, when released, should stay inflated and presents no collision problem with the descending balloon/BVS system. The parachute is released from the balloon by a pyrotechnic device just above the balloon top.

Flotation System Operation

The following sequence of events describes the flotation system operations:

1) BVS and inflation system descend on main parachute at a dynamic pressure of 1 psf;
2) Baroswitch is triggered at preset pressure altitude and starts deployment sequence (about 612 mb pressure);
3) Balloon canister lid is released by operation of two explosive nuts;
4) Parachute extracts balloon by applying load through the diffuser sock and line;
5) Sequencer opens inflation system low-flow start valve releasing gas that extends and fills diffuser sock. At this time the balloon pressure vent system is closed off because it is located in the canister where inflation pressures are relatively high;
6) 2 sec later the sequencer signals the high-flow start valve to open and the gas storage tanks blow down in about 45 sec, inflating the balloon;
7) System shutoff valve is closed;
8) Line is severed between canister and pressurization system by operation of tube cutter;
9) Pressurization system is separated from BVS and drops away;
10) At completion of the inflation system sequence, the balloon overpressure relief system is armed;
11) Parachute is released from balloon;
12) Inflation gas warms up to environmental conditions, causing the balloon to rise to design float density altitude. Overpressure relief valve vents off excess gas to limit superpressure to 6 mb;

13) Makeup system (if carried) is activated as required by command to correct for pressure loss due to such occurrences as crossing the terminator, or encountering large perturbations in the environment.

The nominal dynamic trajectory/time history of the BVS during balloon deployment, inflation, and stabilization is shown in figure 88. At 1600 sec and after altitude stabilization, the effect of releasing a 5-lb drop sonde is shown.

Figure 89 presents the temperature/time histories for the balloon skin (wall), the balloon gas, and the ambient atmosphere. The gas temperature drops to about 405°R (-55°F) during high-pressure tank blowdown, but rapidly heats up due to both balloon wall temperature and radiation inputs. The final supertemperature is about 31°F. Figure 90 depicts the vent system operation. At about 480 sec, the gas has heated and expanded to the point where it begins to vent the excess overboard. During venting, the superpressure is held very near the design value of 6 mb. Dips in superpressure below 6 mb are caused when the BVS oscillates below the design altitude.

The following results of off-nominal conditions show that the basic system is capable of successful operation under most off-nominal conditions, that the system is highly sensitive to some perturbations, and that tradeoffs are possible in many areas where sensitivity to perturbations is not critical.

Inflation time.- The time required to inflate the balloon is controlled by orifice. The nominal orifice size of 0.065 sq in. produces a 45-sec blowdown. A smaller orifice of 0.014 sq in. and a larger orifice of 0.080 sq in. produced blowdown times of 200 and 30 sec, respectively. The altitude traces for the three inflation times show, in figure 91, that inflation time less than 45 sec has little effect on the balloon trajectory. It is apparent also that inflation times as long as 200 sec are not disastrous, but only double the distance the balloon undershoots the float altitude. Temperature profiles for the 200-sec inflation show, in figure 92, that the balloon temperature after inflation drops to 440°R only compared with 402°R in the nominal inflation. The balloon penetrates the atmosphere to a higher temperature, but this effect is not appreciable (555 vs 540°R in the nominal case).

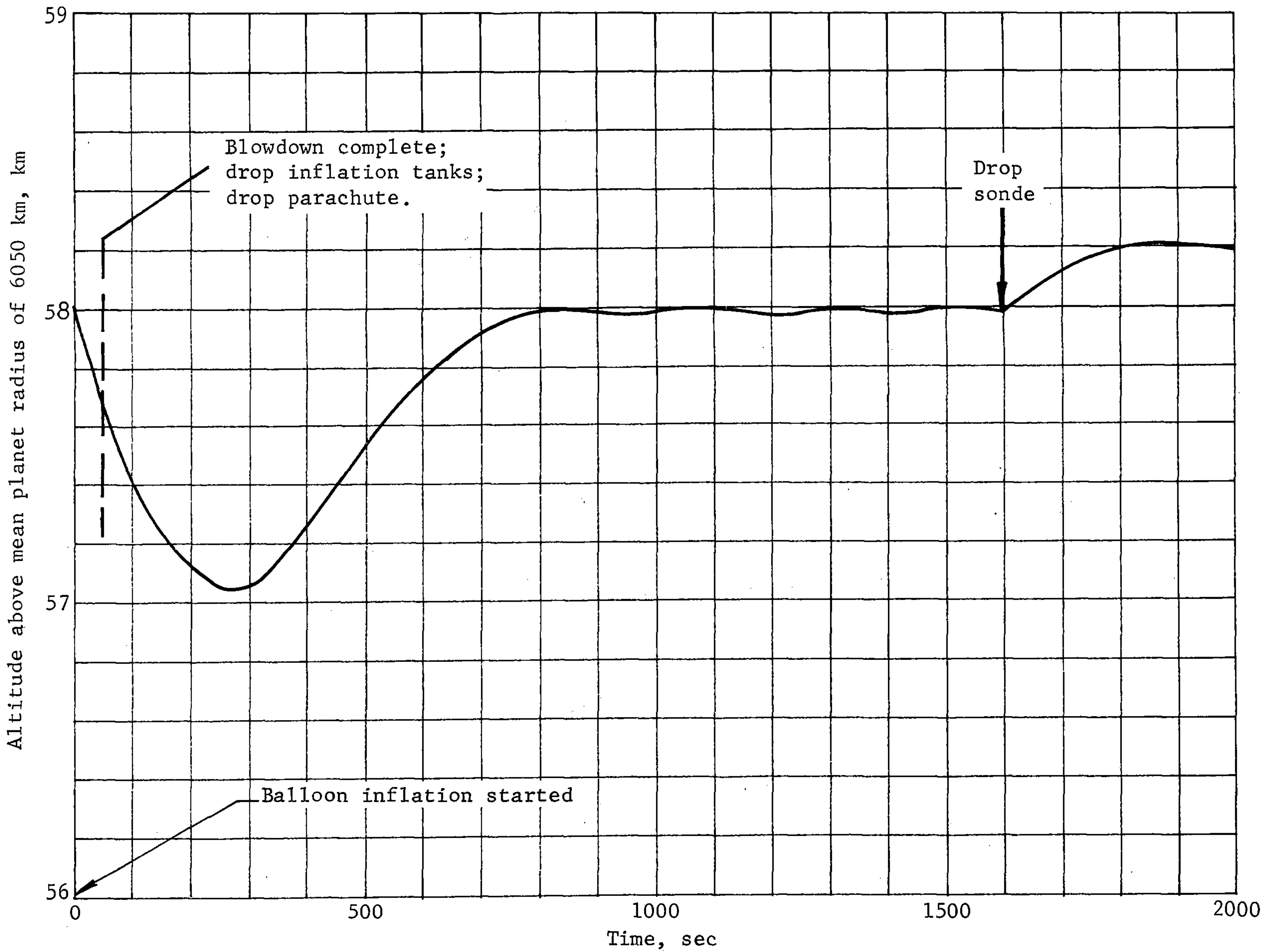


Figure 88.- Balloon Altitude Profile

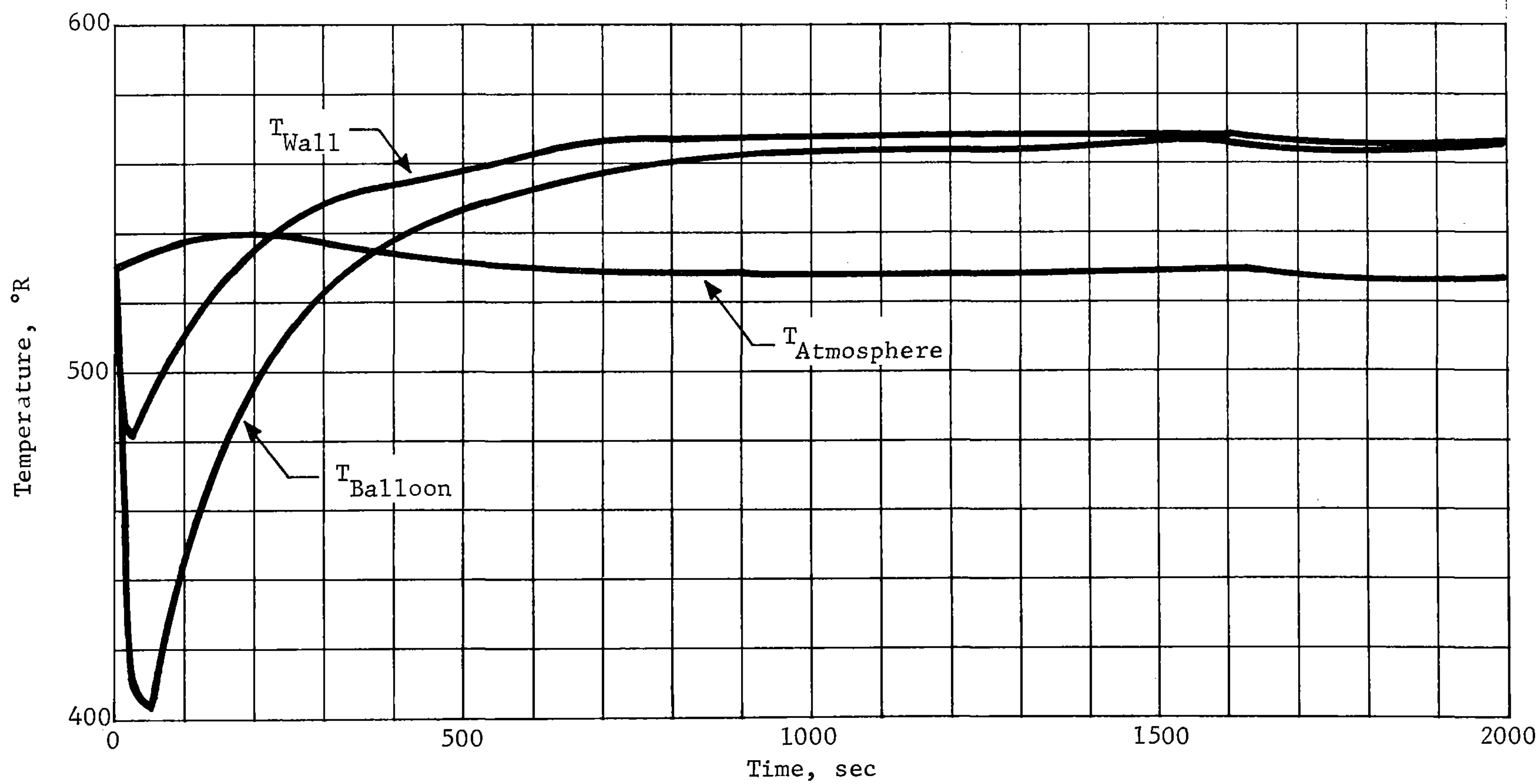


Figure 89.- Temperature Profile

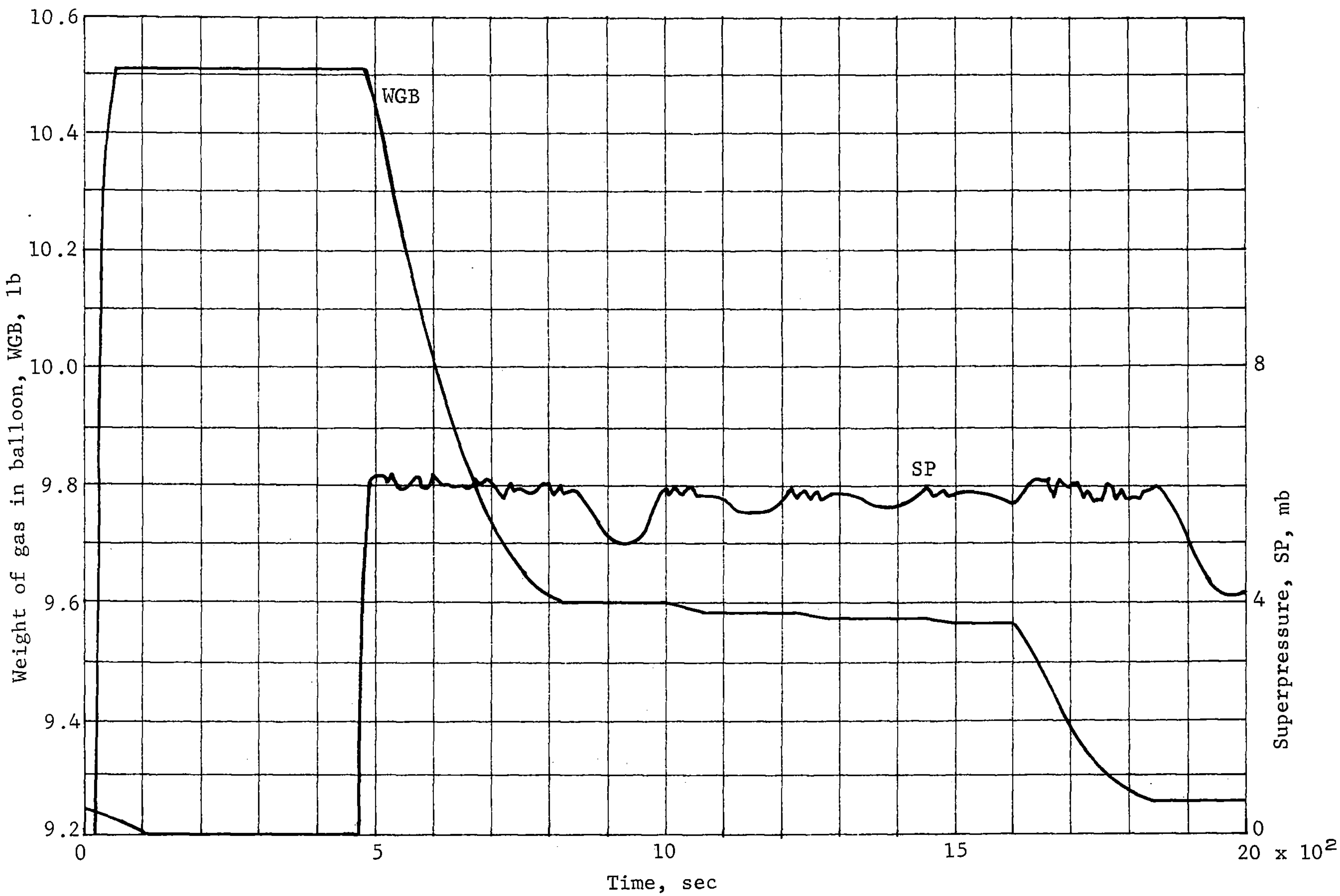


Figure 90.- Gas Weight and Balloon Superpressure

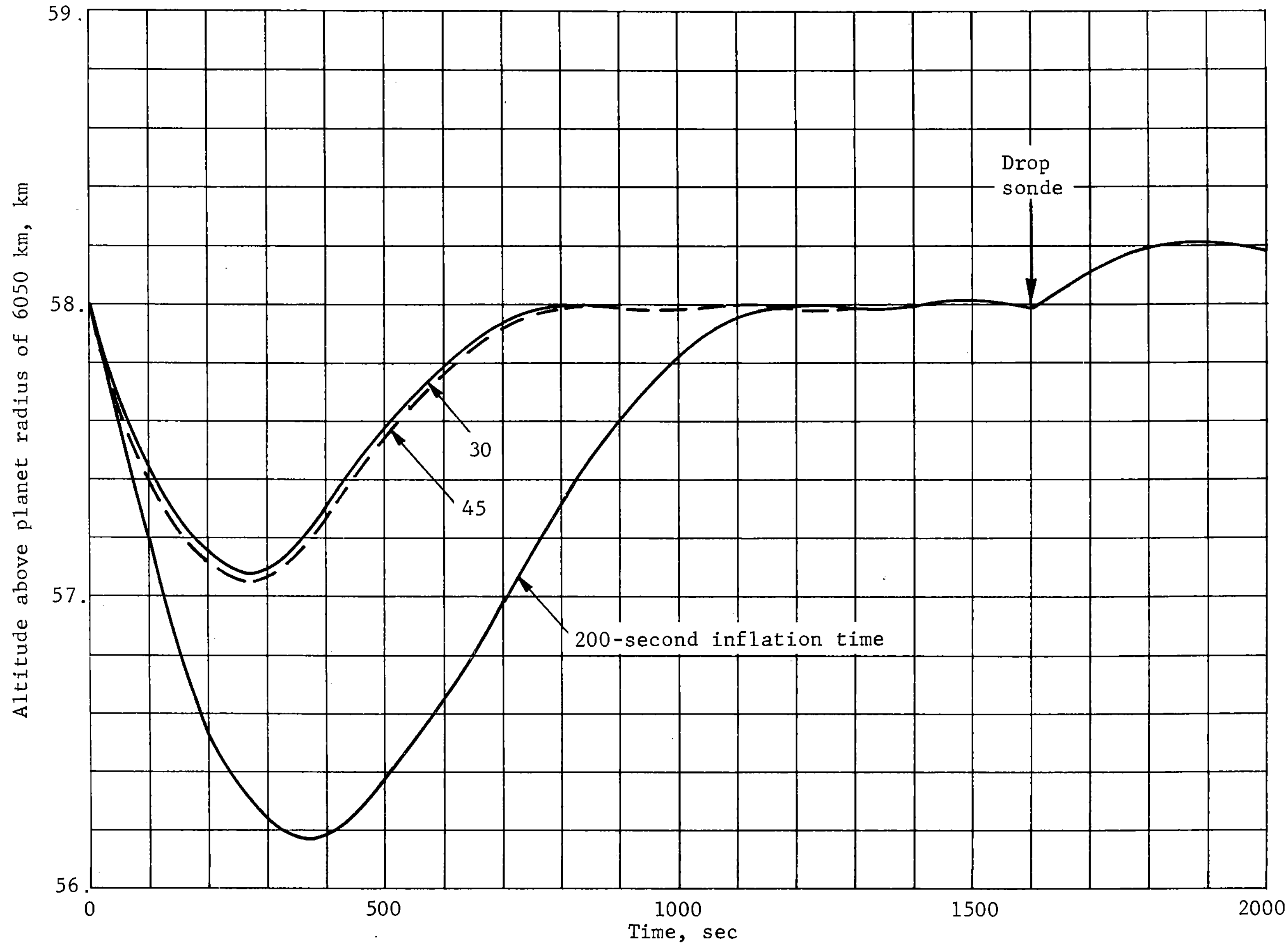


Figure 91.- Variation of Inflation Time

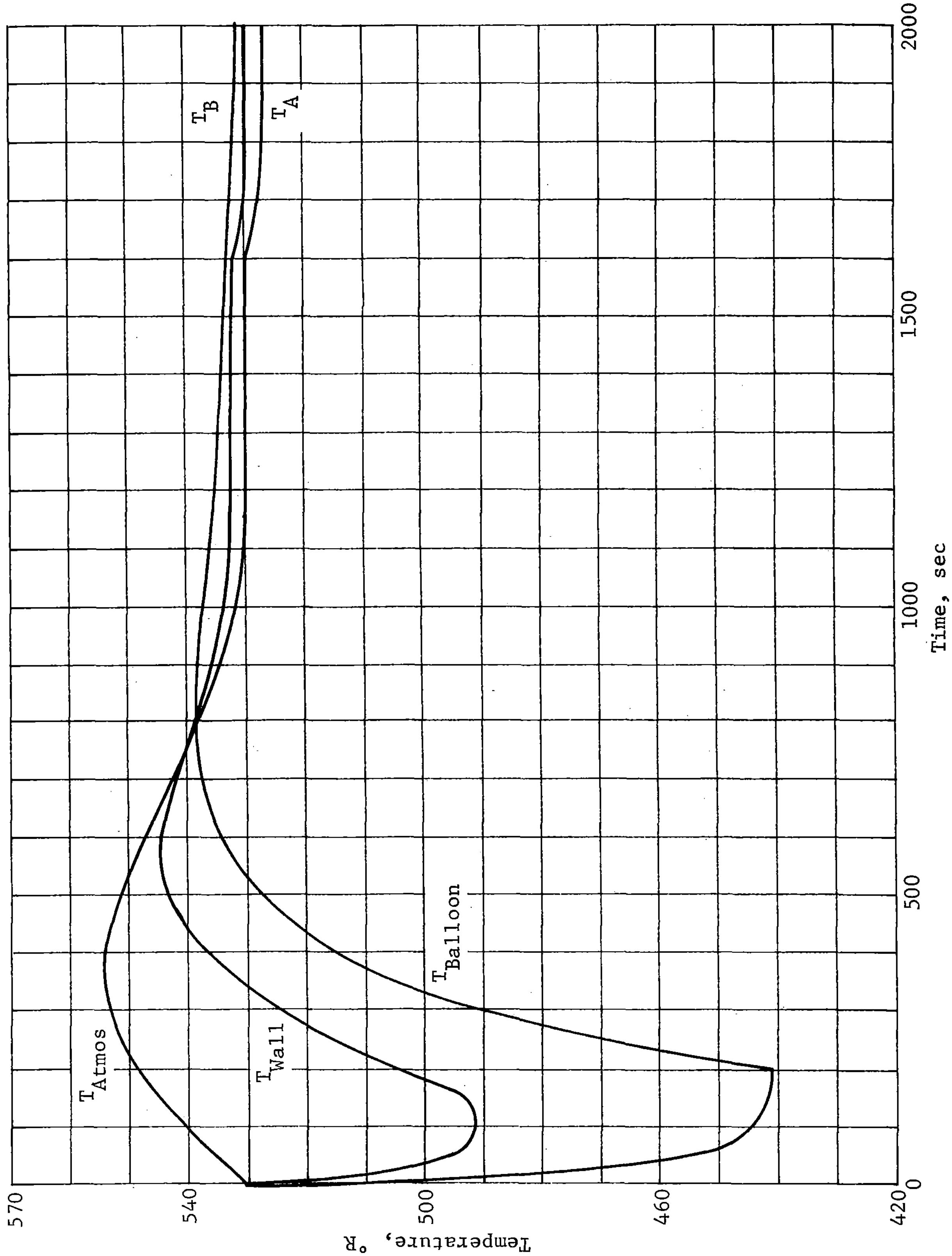


Figure 92.- Temperature Profiles for 300-sec Inflation

Deployment altitude.- The basic system design uses a barometric pressure sensing device to initiate balloon inflation. Typically, the tolerance on a barometric sensing device is considerably less than 20 mb. A value of 20 mb, which translates into ±600 ft of altitude, is used in the study. The effect of a variation in deployment altitude of this magnitude is seen in figure 93 to cause no difficulty in achieving the desired float altitude. The largest deviation from the nominal trajectory is nowhere greater than 0.2 km.

Atmosphere variation.- This system has no difficulty achieving a float condition in the upper and lower Venus atmospheres (appendix A) as seen in figure 94. Because of the different density profiles the balloon floats approximately 0.2 km above and 0.32 km below the mean atmosphere float altitude.

In and out of clouds.- The present BVS concept involves balloon inflation below the clouds of Venus, which are assumed to completely cover the planet below 70 km. During the nominal mission, the balloon never rises above the postulated cloud top and, therefore, doesn't encounter the changes in balloon temperature that would accompany in-and-out of cloud operation. If one speculates about the character of the Venus clouds on the basis of Earth cloud formations, the possibility of an irregular cloud top might be considered to produce in-and-out of cloud conditions as the balloon floats along at constant density. Certainly, there are no data at present to support the intermittent or irregular cloud cover supposition. However, this possibility raises a potential threat to the BVS system.

To study the above problem, a hypothetical situation was assumed in which inflation and initial float occur in the clouds as discussed above, followed by a 10-minute out-of-clouds period and subsequent return into the clouds. In figure 95, the temperature of the gas in the balloon is seen to rise 64° as the balloon drifts out of the clouds. Venting of 1.02 lb of hydrogen occurs during this period. When the balloon drifts back into the clouds, the temperature drops, and the balloon begins to descend as indicated in Figure 96. The balloon no longer has enough free lift and will continue this descent to the planet surface unless makeup gas is added or ballast dropped.

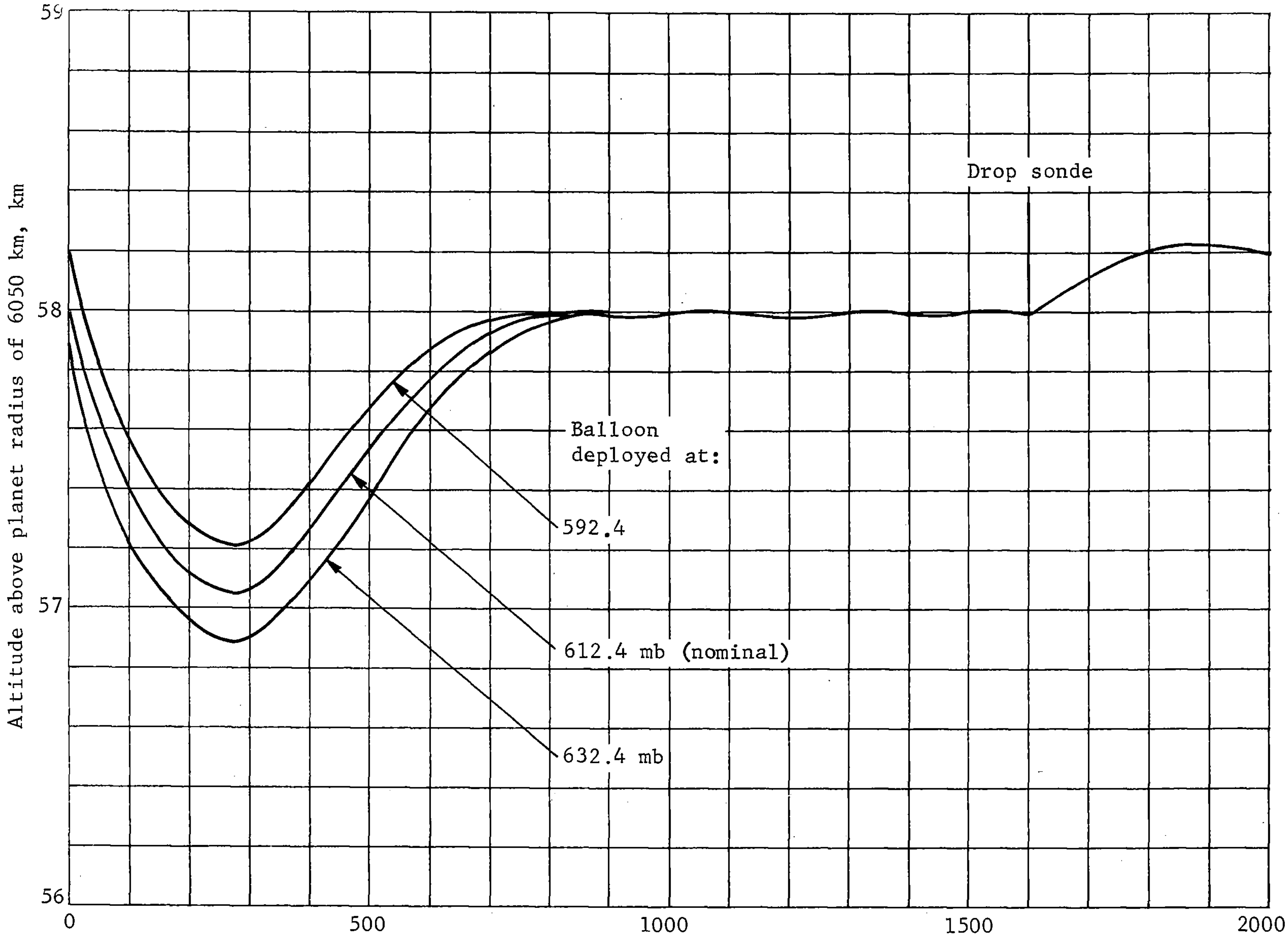


Figure 93.- Variation of Deployment Altitude

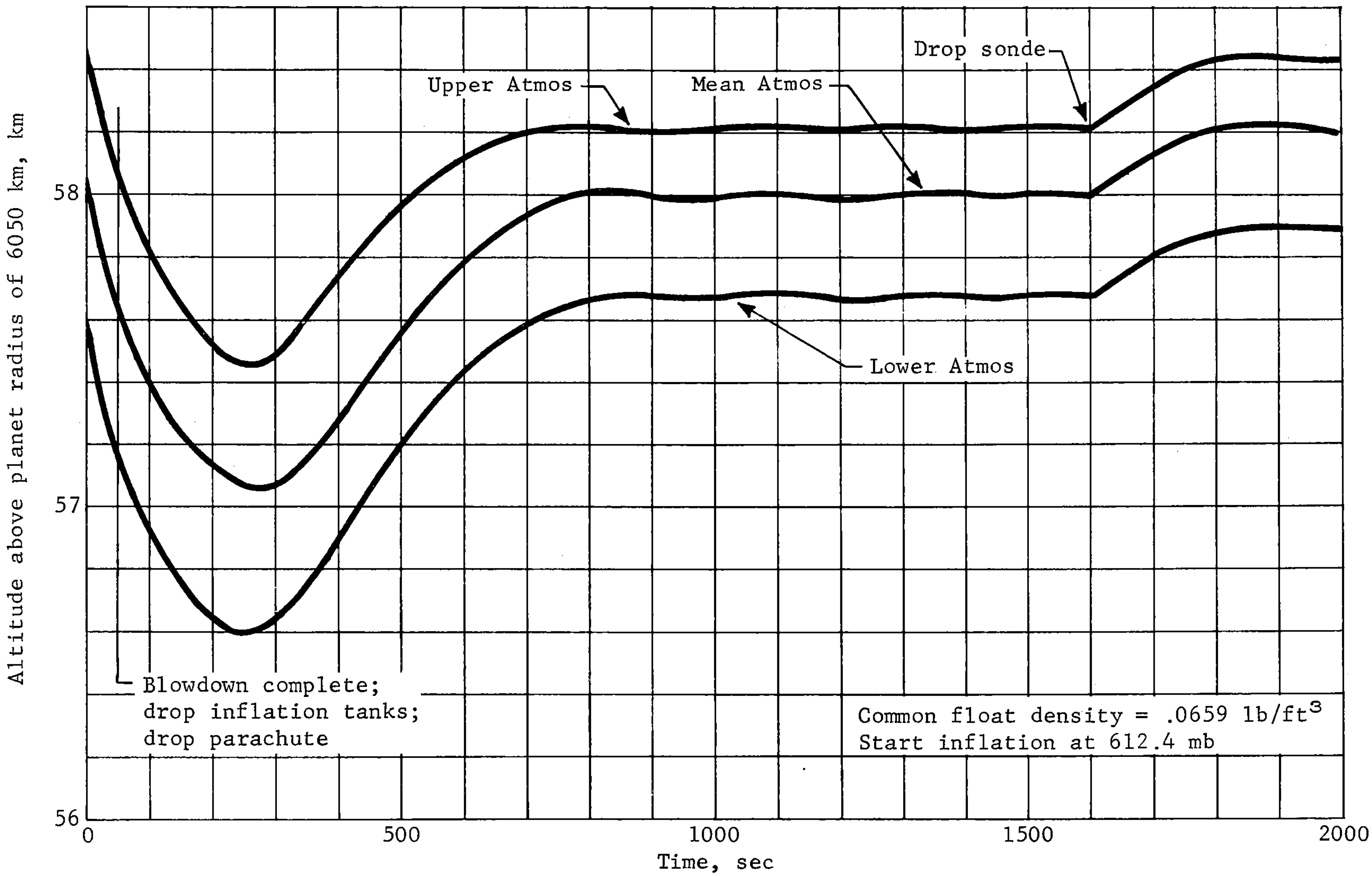


Figure 94.- Nominal System Design in Upper and Lower Atmosphere

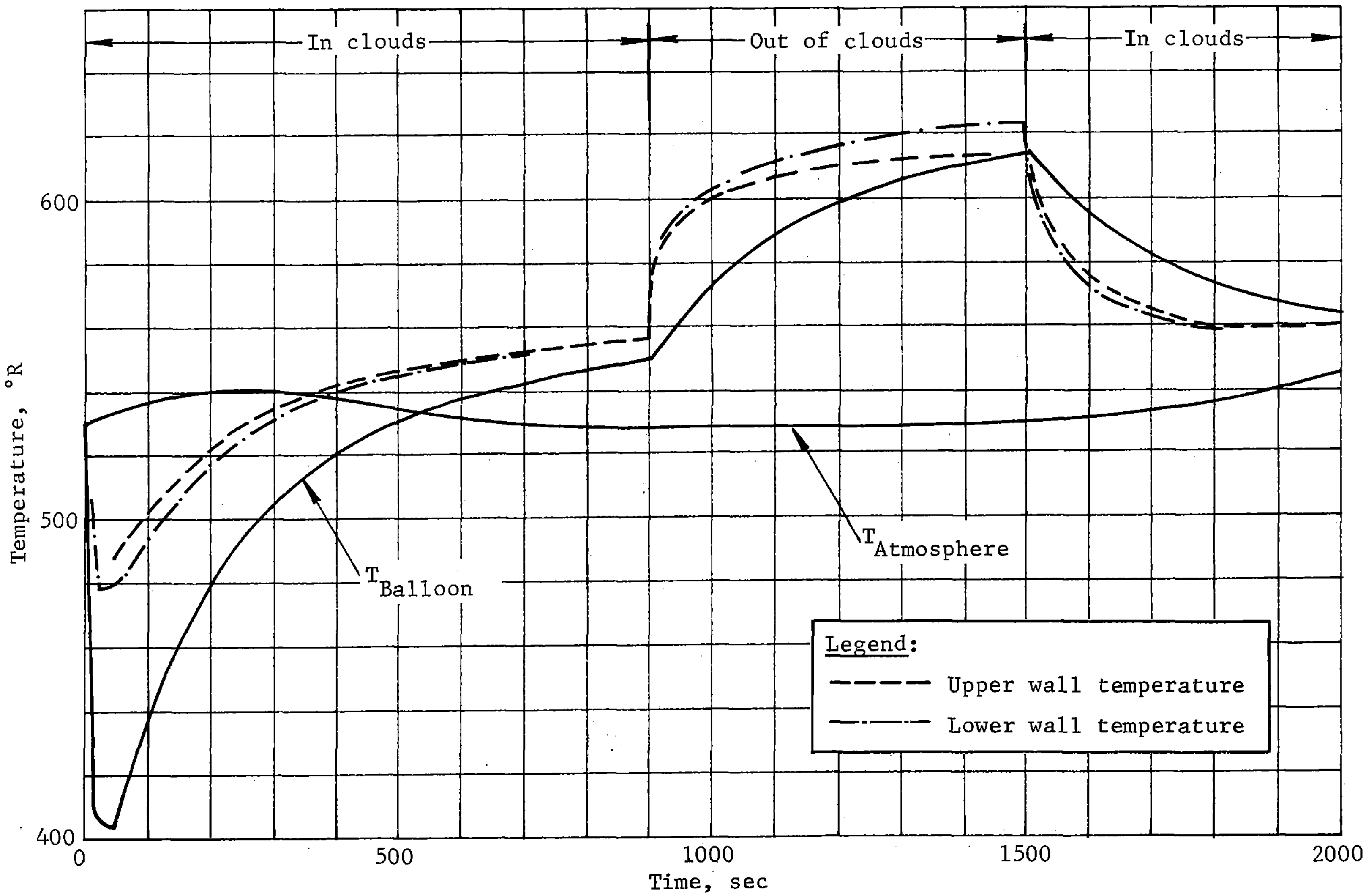


Figure 95.- Temperature Variation in and out of Clouds

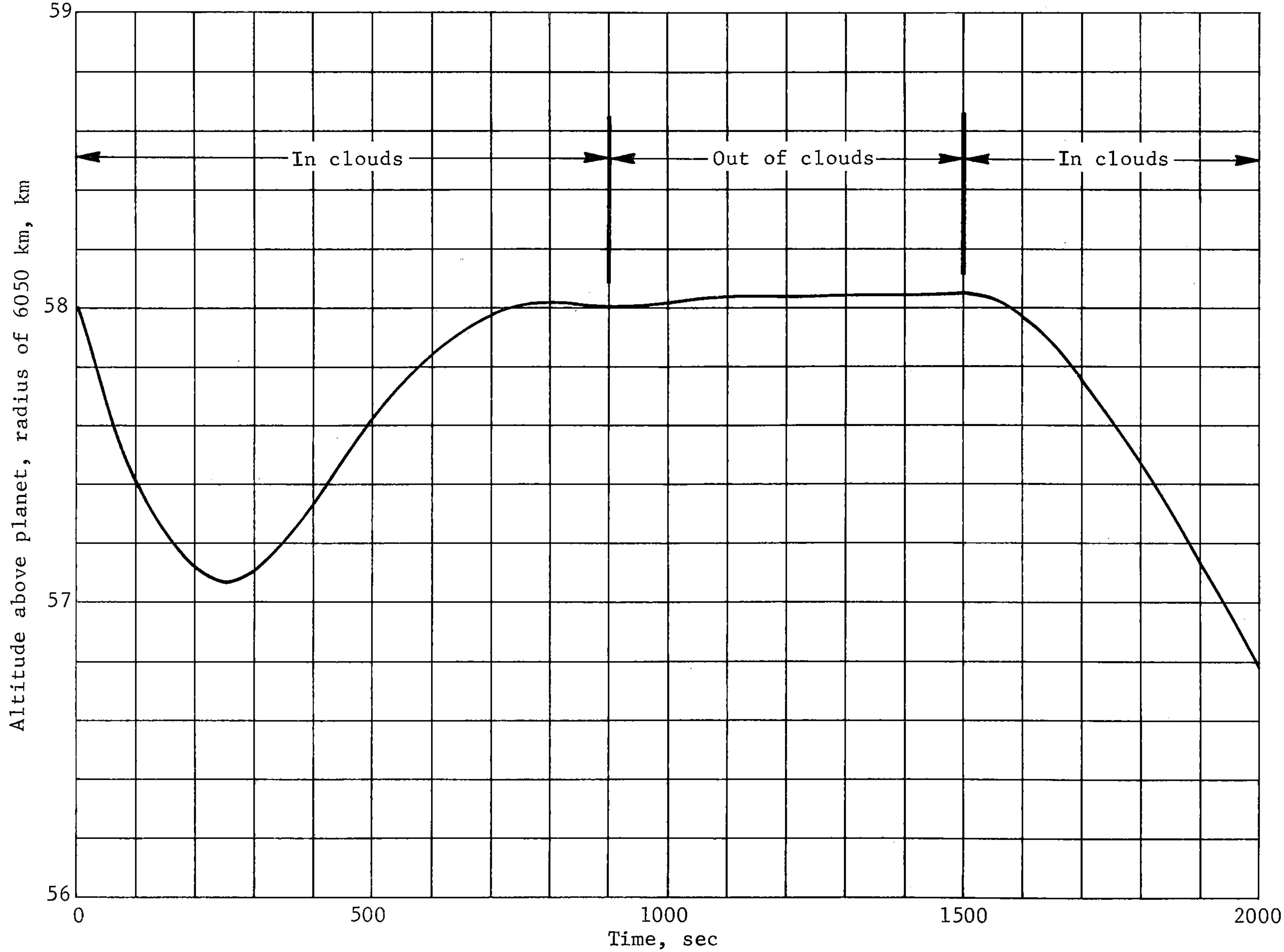


Figure 96.- Reaction in and out of Clouds

Terminator.- The BVS requires special control hardware to accommodate the loss in the balloon supertemperature when the BVS crosses the terminator into the dark side of Venus. It is assumed that the balloon remains in the clouds as it crosses the terminator, and, when the solar radiation drops to zero, the balloon cools down to ambient temperature. The ambient temperature, pressure, and density are assumed to remain the same on either side of the terminator within the clouds. This cooling results in a nominal 25°F balloon temperature loss with a corresponding theoretical 27.5-mb pressure loss. However, after 5.4°F cooling, the balloon reaches zero superpressure, and the volume then reduces by about 110 ft^3 during further cooling. The volume reduction results in negative free lift, and the balloon will descend to the surface unless corrective action is taken. It can be shown that the free lift for a zero pressure balloon is:

$$\text{FREE LIFT} = W_G \frac{M_A T_G}{M_G T_A} - W_{LOFT}$$

where

W_{LOFT} = total weight

W_G = weight of gas

M_A/M_G = molecular weight ratio, air to gas

T_G/T_A = temperature ratio, gas to air (this value approaches 1)

or, in other words, once the zero pressure balloon begins descent, the free lift (in this case negative) is a constant and the balloon will descend to the surface.

Corrective action can be in the form of either dropping weight or adding makeup gas. A weight decrease of 8.1 lb is required. Release of drop sondes can meet this requirement. If a gas makeup system is used, it will require a minimum of 0.45 lb of hydrogen gas at 4500 psi in a 6.2 lb, 10.2-in. diameter tank. Special control hardware is required to sense the loss in superpressure and release the drop sonde(s) or makeup gas at the appropriate time. If either is done too early, the balloon may rise and vent before it has completely cooled.

Vertical winds.- A parametric curve is presented in figure 97 to demonstrate the sensitivity of the balloon system to updrafts. During an updraft, the balloon will always remain fully inflated because it will remain in a superpressure condition. Therefore, its volume and drag cross section remain constant and, for a steady-state updraft, the balance of forces is:

Weight lofted = buoyancy force + drag force

$$Wt_{LOFT} = \rho_A V_B + \tfrac{1}{2} C_D \rho_A \frac{V^2 A}{g}$$

where

ρ_A = density of atmosphere, lb/ft^3

V_B = volume of balloon, ft^3

C_D = drag coefficient

V = updraft velocity

A = cross section area of the balloon

g = gravitational acceleration

It can be seen that the balloon will rise into the lower density atmosphere until the buoyancy force reduces enough to equal the drag force due to the vertical wind. As an example, in figure 97, an updraft of 8 fps (a purely hypothetical case) causes the balloon to rise 0.31 km above the nominal float radius of 6108 km. The ambient pressure reduction produces an effective superpressure in the balloon of 29 mb. This exceeds the nominal design superpressure of 6 mb and will cause venting to occur. The problem occurs when such an updraft ceases allowing the balloon to begin to return to its original altitude. However, some of its gas has been vented and the balloon no longer has enough gas to produce free lift.

The results of dynamic simulation of the above updraft problem are shown in figure 98, where an 8-fps updraft is seen to cause the balloon to rise 0.34 km above the nominal float altitude and approaches an equilibrium condition when the updraft is instantaneously removed. As expected, the balloon vents 0.44 lb of hydrogen during the updraft and cannot sustain its original float condition when the gust is removed. The balloon continues its descent to the planet surface unless corrective action is taken.

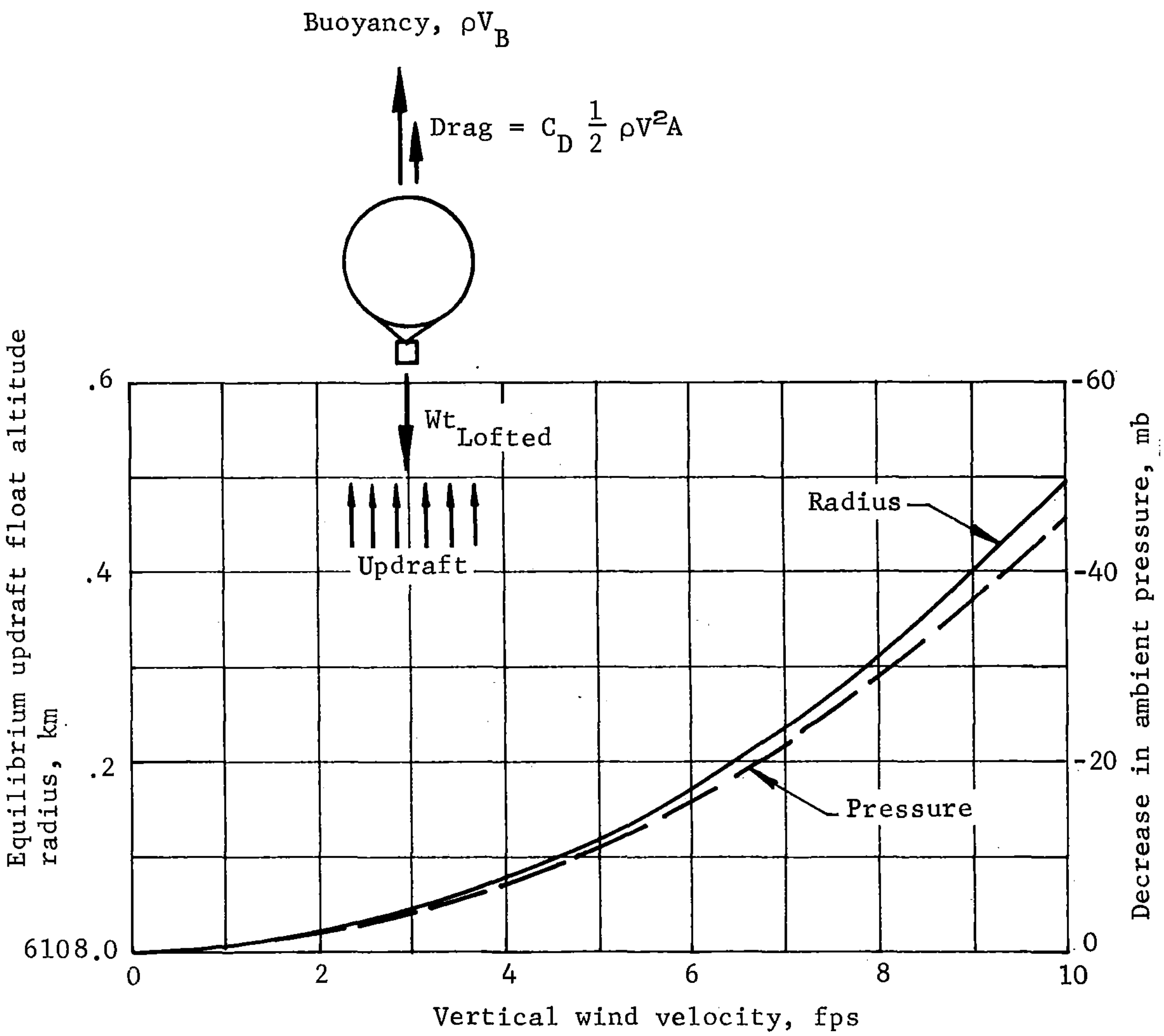


Figure 97.- Effects of Updraft on Performance

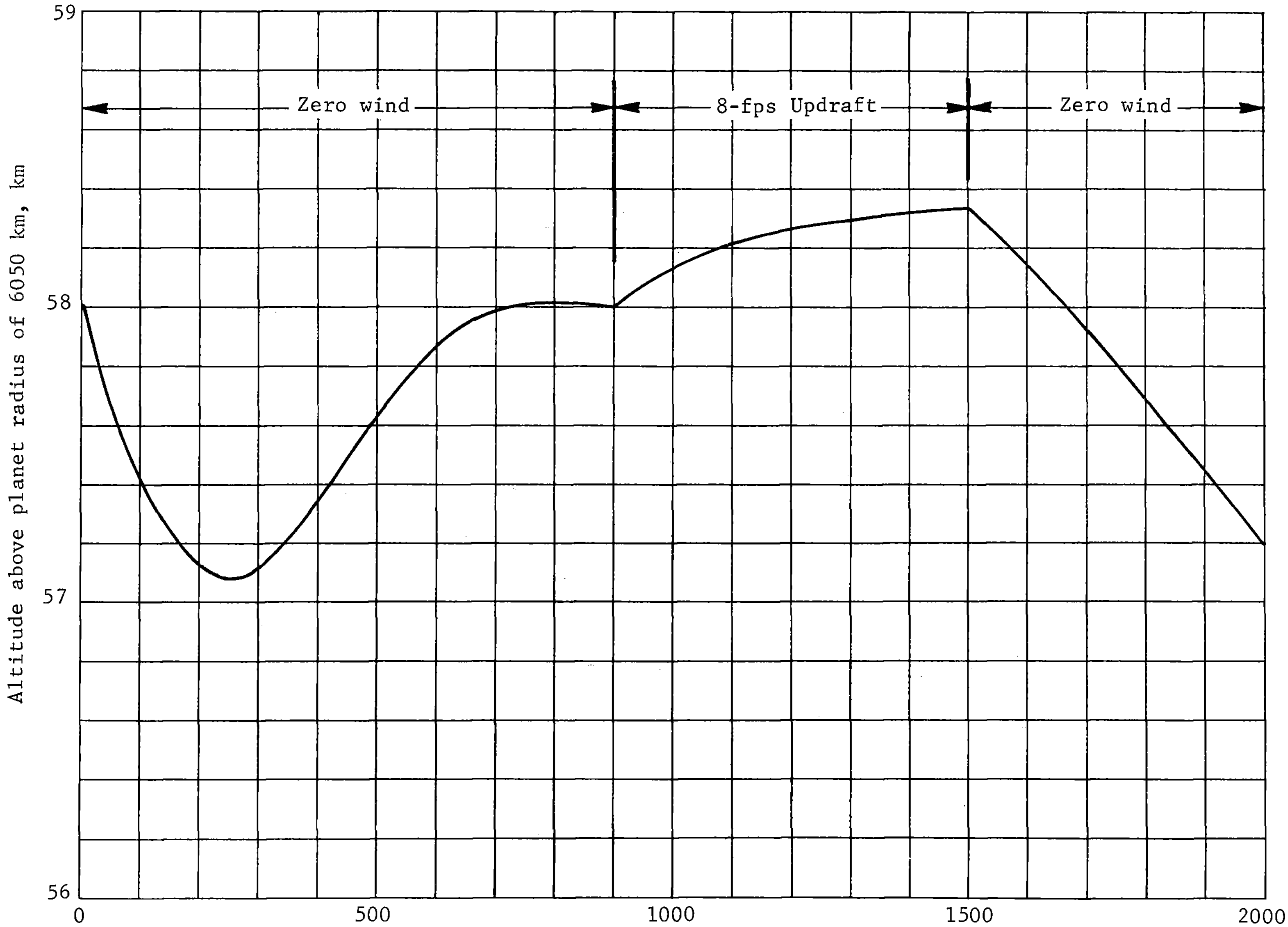


Figure 98.- System Reaction to 8-fps Vertical Updraft

On the basis of figure 97, a 6-mb superpressure balloon can withstand a vertical wind of 3.5 fps without encountering the above problem. Turbulence is critical to the BVS concept but is not likely to be very accurately known before the first BVS mission.

Initial pressurant temperature.- An assumption is made in all of the nominal balloon deployment cases that the pressurant in the blowdown tanks is at ambient atmospheric temperature (529.8°R at deployment altitude in the mean Venus atmosphere). This is not an unreasonable assumption because temperatures above and below this value occur in the atmosphere before balloon inflation. The uncertainty in the extent of heat transfer through the tank walls requires that a tolerance on the pressurant temperature be examined. A lower than nominal temperature before inflation aggravates the temperature drop during inflation. A temperature of 400°R, which is lower than the lowest atmosphere temperature occuring anytime during entry, is compared in figure 99 with the nominal case. The low-pressurant temperature increases the time required to reach equilibrium float by about 3 minutes and undershoots the float altitude by an additional 0.4 km. Neither of these results is considered a problem.

Dark side deployment.- Balloon deployment on the 1973 Venus/Mercury swingby mission will occur in the clouds, but 30° from the terminator on the dark side of the planet. The effect of a slower balloon temperature rise and lower supertemperature shown in figure 100 produces the trajectory profile shown in figure 101. No difficulty is experienced in achieving float altitude. If the balloon does cross the terminator into the light side due to winds, the balloon will vent gas as required and will continue to float essentially at the desired altitude. Should the balloon subsequently be blown back across the terminator, the problem of crossing the terminator from light to dark side will occur and require a solution.

Balloon thermal analysis.- A study was conducted to determine the balloon gas and wall temperatures at nominal conditions for both in and out of the clouds. The parameter values associated with the nominal conditions are listed in table 79. Although the cloud tops are expected to be above float altitude, a break in the cloud layer is not unlikely. Additionally, temperatures were calculated for the extreme environments (table 79) to which the balloon may be subjected.

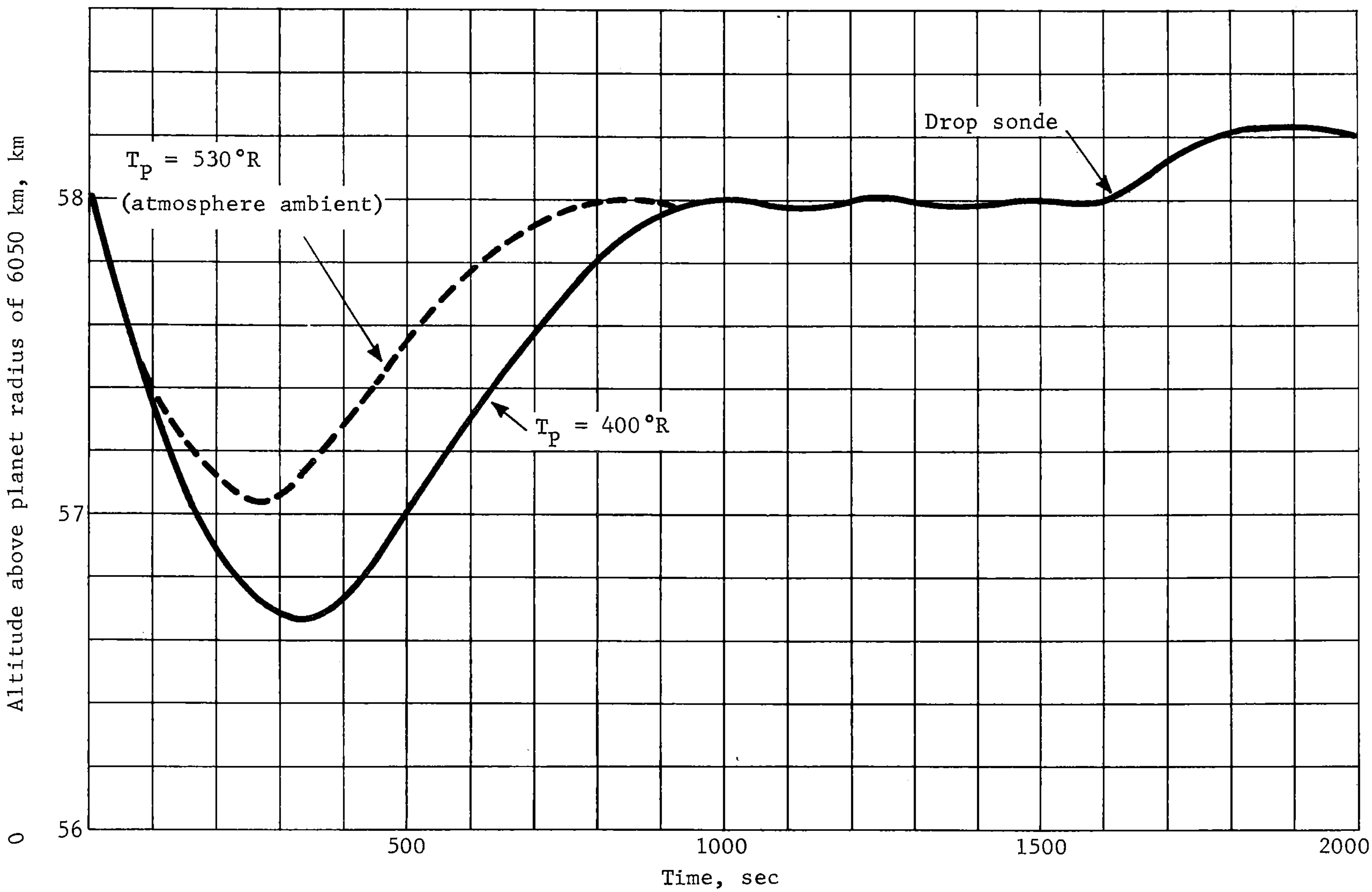


Figure 99.- Variation of Pressurant Temperature at Inflation

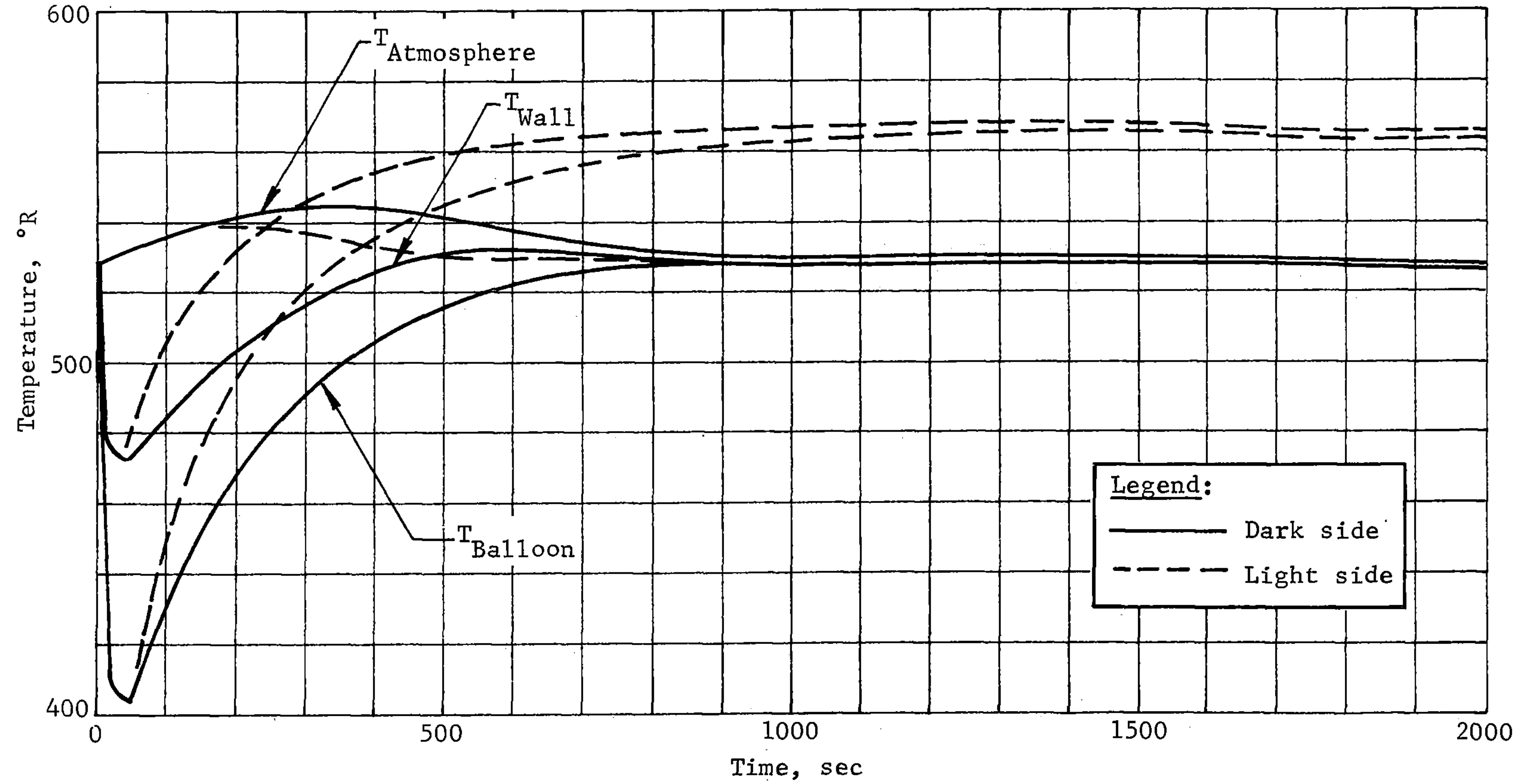


Figure 100.- Temperature Profile, Dark Side Deployment

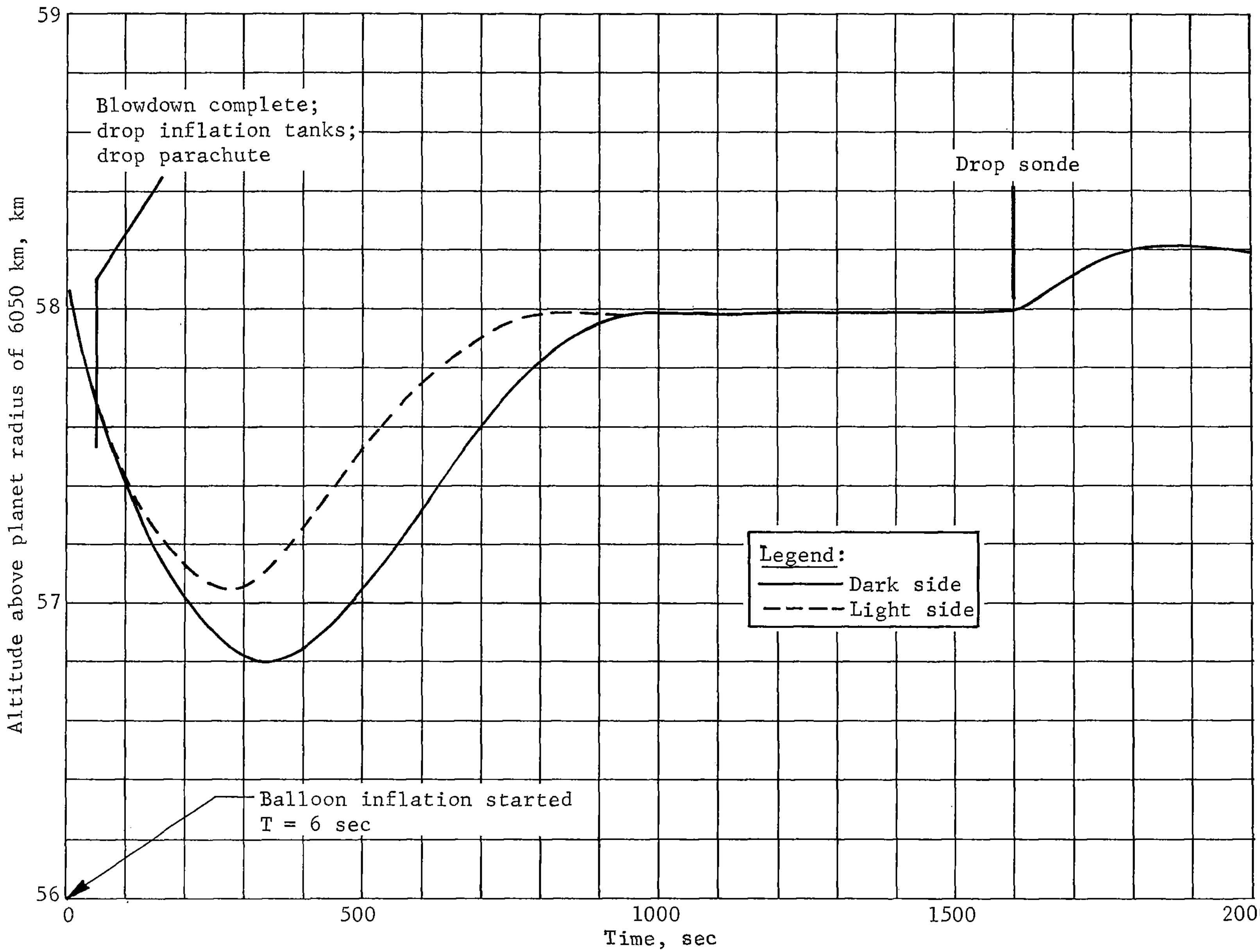


Figure 101.- Altitude/Time History, Dark Side Deployment

TABLE 79.- BALLOON PARAMETERS

Parameter	Range of values	Nominal conditions		Cold extreme	Hot extreme
		In clouds	Out of clouds		
Atmosphere, % CO_2/% N_2	85/15 to 95/05	90/10		85/15	95/05
Float altitude, ft	190 300 $^{+1310}_{-1640}$	190 300		188 660 (t_a = 59°F)	191 610 (t_a = 84°F)
Cloud top altitude, ft	210 000 ± 60 000	>190 300	<190 300	<188 660	<191 610
Cloud top temperature, °F	-22 ± 27	-22		-49	5
Atmospheric temperature, t_a, °F	Function of atmosphere and altitude, ±9	70		50	93
Sun angle, deg	0 to 90	0		90	0
Cloud solar transmissivity	0 to 1.00	.14		----	----
Cloud emissivity	1.00	1.00		----	----
Atmospheric transmissivity	.85 to 1.00	.93		.85	1.00
Atmospheric emissivity	.20 to 1.00	.25		.20	1.00
Albedo	.50 to .90	.76		.50	.90
Balloon solar absorptivity, α_s	[a].05 to .20	.10		.05	.20
Balloon solar transmissivity, τ_s	.90 to .75	.85		.88	.77
Balloon infrared absorptivity, ϵ	[b].05 to .20	.10		.05	.20
Balloon infrared transmissivity, τ_{IR}	.90 to .75	.85		.85	.80
Multiplier in heat transfer coefficient equations	1.0 ± 50%	1.0		1.5	.5

[a] $\alpha_s + \tau_s = 0.95 \pm 0.02$

[b] $\epsilon + \tau_{IR} = 0.95 \pm 0.05$

Note: Balloon properties are based on 1.5 mil thickness; variation with thickness may be neglected as long as the thickness does not vary more than 1.0 mil.

The analytical model which was used in this study considered that the balloon gas, H_2, radiates no heat and is perfectly transparent to foreign thermal radiation. The balloon material absorbs radiation from two primary sources -- the sun (direct and albedo) and Venus and its clouds and atmosphere (infrared). The material emits infrared radiation. Multiple reflections of solar and infrared radiation inside the balloon were considered. Additional heat transfer from the material takes place through convection to the environment and to the hydrogen. The developed computer model considers the balloon to consist of four isothermal, equal area, segments. A schematic of the balloon and its environment is shown in figure 102.

The following assumptions were made in this analysis:

1) Albedo is reflected from the top of the clouds and is diffused;
2) Clouds emit infrared radiation as a black body;
3) Infrared radiation is emitted from the atmosphere at the local atmosphere temperature;
4) Solar and infrared reflections inside the balloon are diffused;
5) Solar spectrum does not change due to absorption of various wavelengths.

The balloon gas steady-state temperatures for the nominal and extreme conditions are shown in table 80. The temperature excursions that will occur as the balloon floats in and out of the clouds are 59 and 12°F for the sun and dark sides, respectively. On the light side, the hot extreme occurs when the sun is directly overhead ($\phi = 0$), and the cold extreme occurs when the sun is on the horizon ($\phi = 90°$). Wall temperatures generally range between $\pm 10°$ of the gas temperature, with the lower half of the balloon being warmer than the upper half.

Coating the balloon with a conversion coating was investigated at nominal environmental conditions. This coating, which can be applied to the balloon material and withstand the inflation process, is similar to the coatings used on PEGASUS and Echo II. The results showed that the coating increases the supertemperature approximately 5°F both in and out of the clouds.

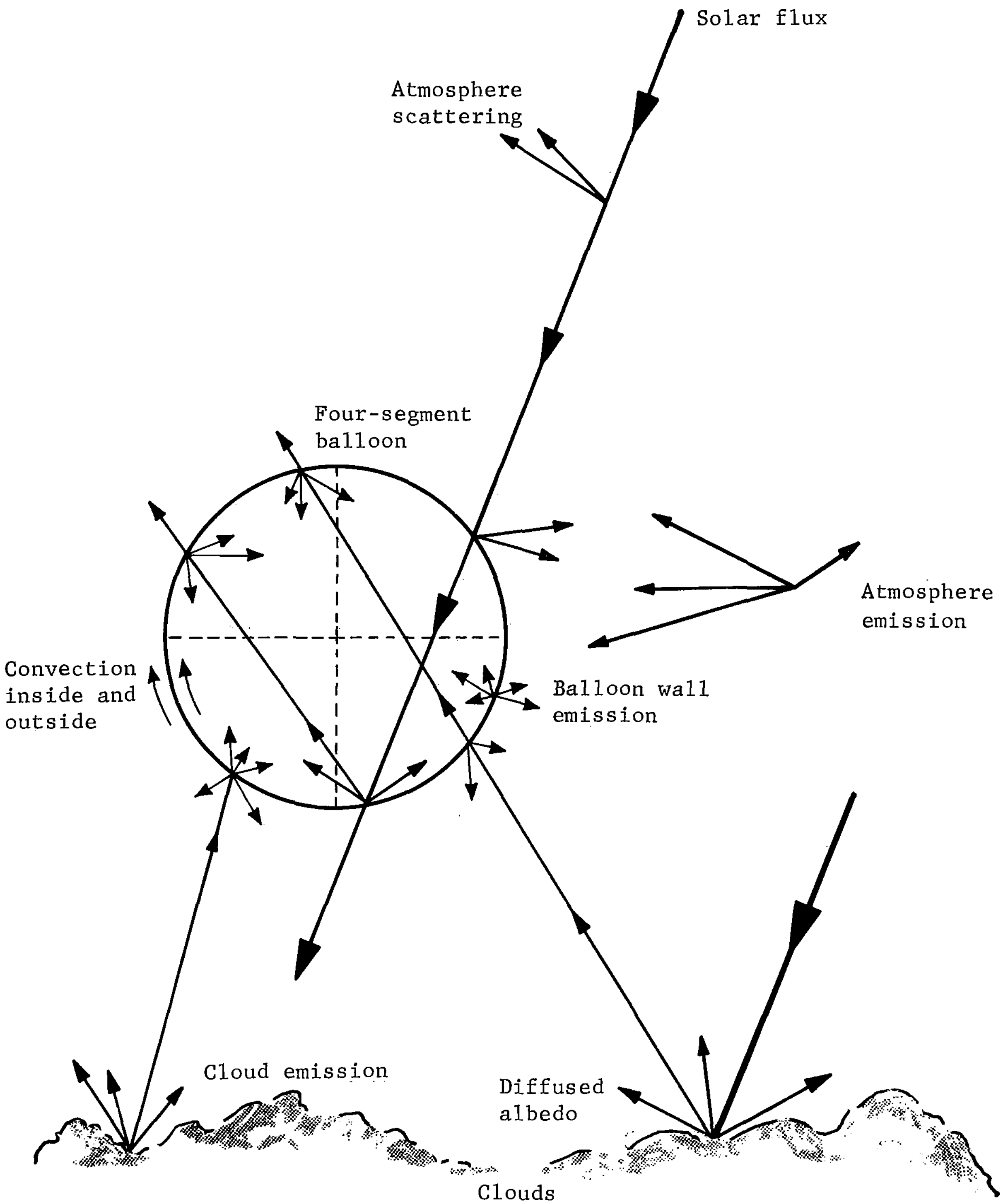


Figure 102.- Balloon and Environment

TABLE 80.- BALLOON GAS TEMPERATURE

Environment	Atmosphere temperatures, °F	Balloon location on planet	Balloon gas temperature, °F	
			In clouds	Out of clouds
Hot	93	Dark side	93	----
		Light side	----	267
Nominal	70	Dark side	70	58
		Light side	101	160
Cold	50	Dark side	----	37
		Light side	----	37

A supplementary study was conducted to determine the sensitivity of the gas temperature to various parameters over their expected ranges. The effects of these parameters are shown in figures 103 through 107. Figure 107(a) combines the effects of the two parameters, sun angle and cloud transmissivity, shown individually in figures 106(a) and (b). These two are interrelated, in that cloud transmissivity is a function of the sun angle.

Certain parameters have a pronounced effect on the gas temperature, whereas others have little or no effect. Those that are most significant are the constant factor in the heat transfer coefficient equation, the balloon material properties and the atmosphere temperature shown in figures 103(a), 103(b), and 104(a), respectively. The sensitivity to some differs between in and out of the clouds, e.g., albedo. The steady-state gas temperature will be higher out of the clouds than in the clouds, except when the sun is near the horizon or when the cloud transmissivity is approximately 0.50 or greater [figs. 106(a) and (b)].

Active thermal control techniques are not feasible, because the balloon is too large. The balloon material should be as transparent as possible to both solar and infrared radiation. Additionally, a coating should not be used.

<u>Balloon stress analysis</u>.*- Wherever possible, the balloon is treated as a pressure vessel. The governing criteria being an 18.05-ft diameter sphere operating at 10 mb internal pressure with a basic membrane stress of 6000 psi. The structure must also have the capability of supporting a suspended static load of 175 lb and a transient load of 500 lb.

The general configuration is shown in figure 108.

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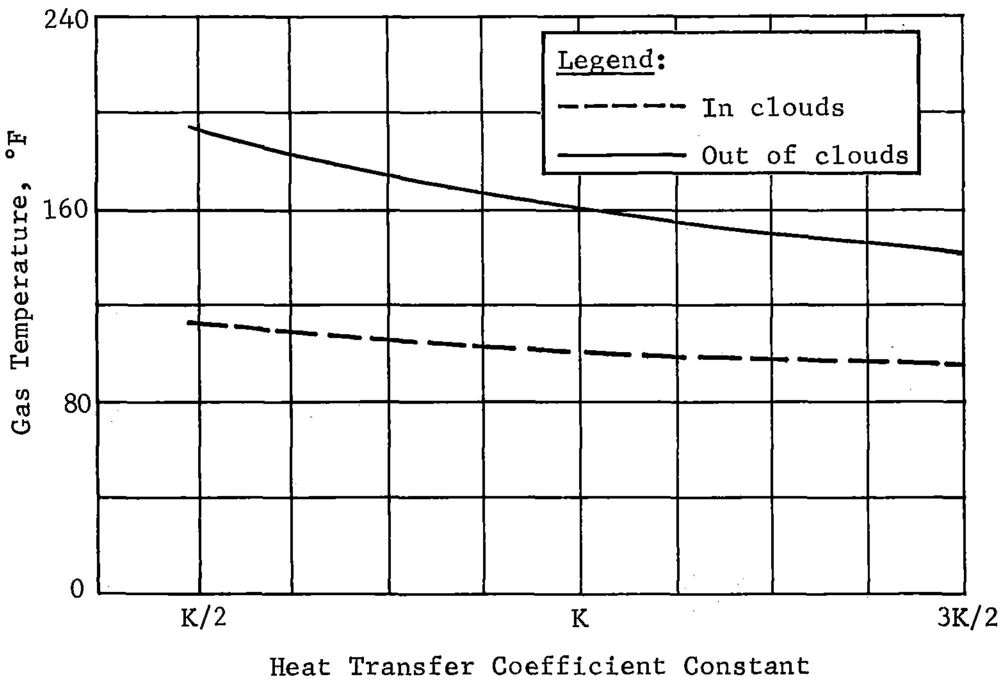
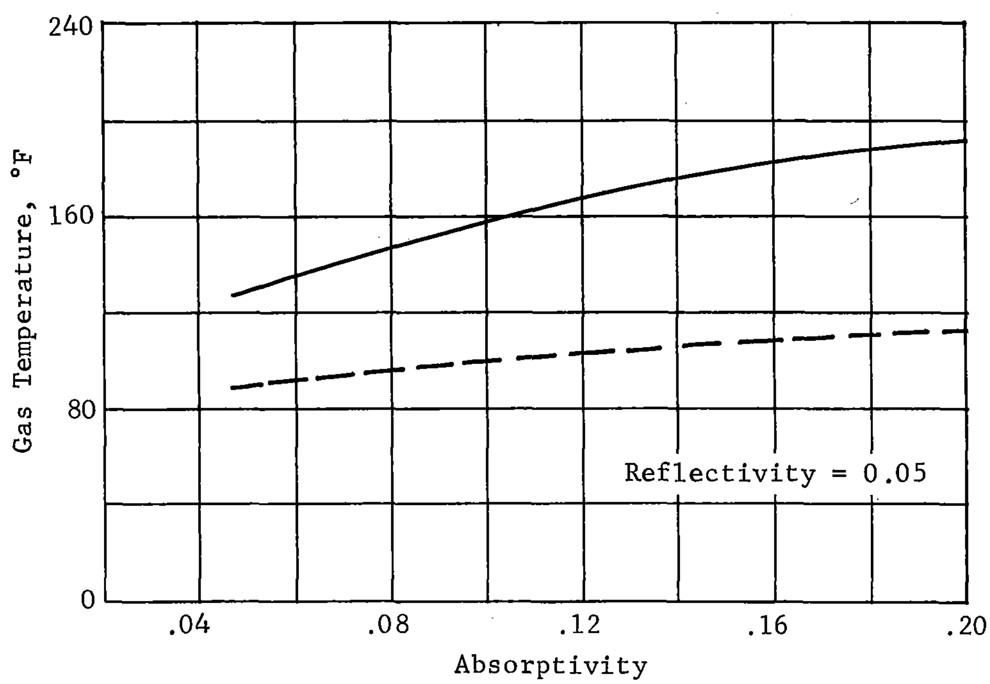
*Raven Industries Inc., Sioux Falls, So. Dak.



(a)



(b)

Figure 103.- Gas Temperature Sensitivity to Heat Transfer and Absorptivity

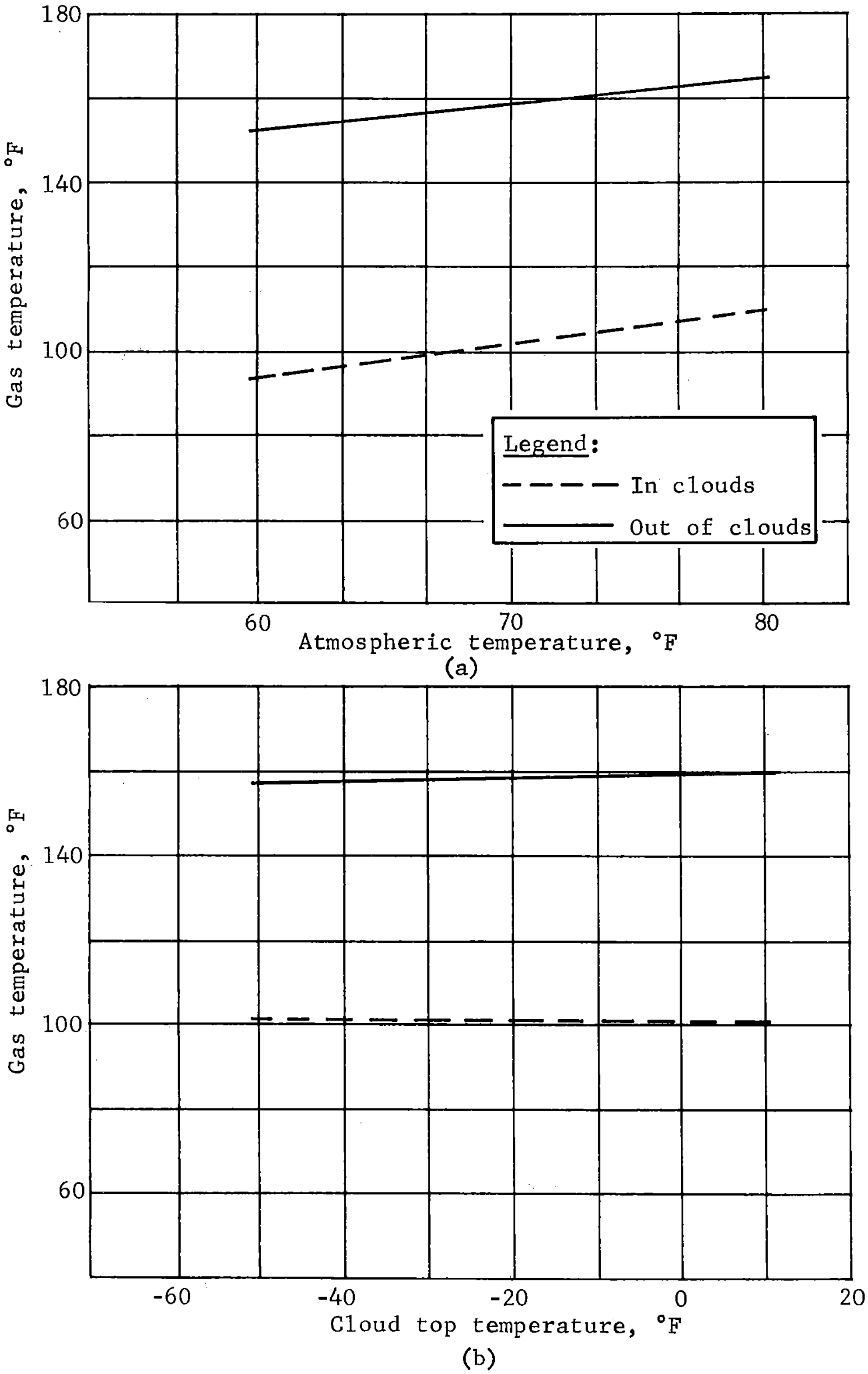


Figure 104.- Gas Temperature Sensitivity to Atmospheric Temperature and Cloud Top Temperature

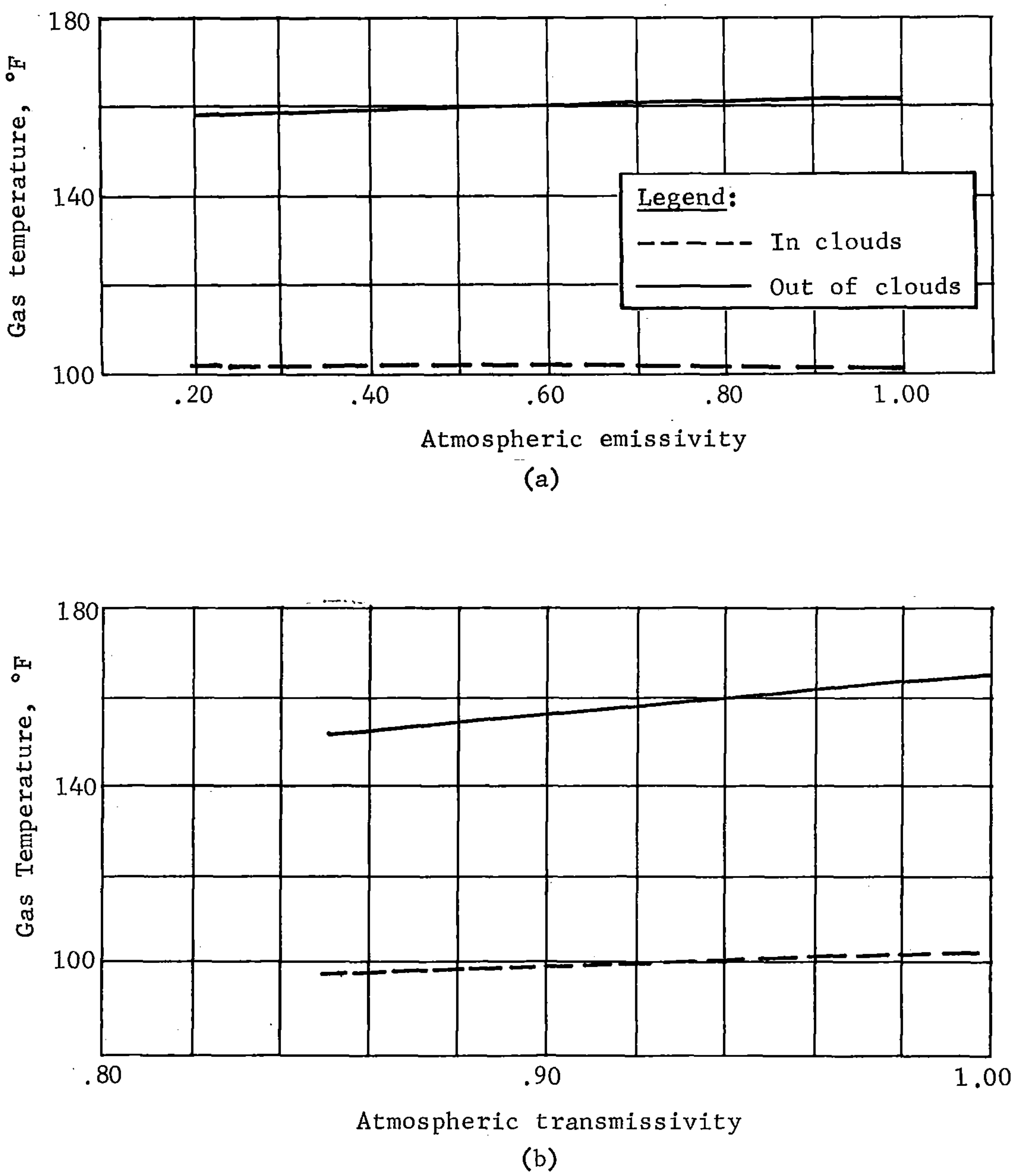


Figure 105.- Gas Temperature Sensitivity to Atmospheric Emissivity and Atmospheric Transmissivity

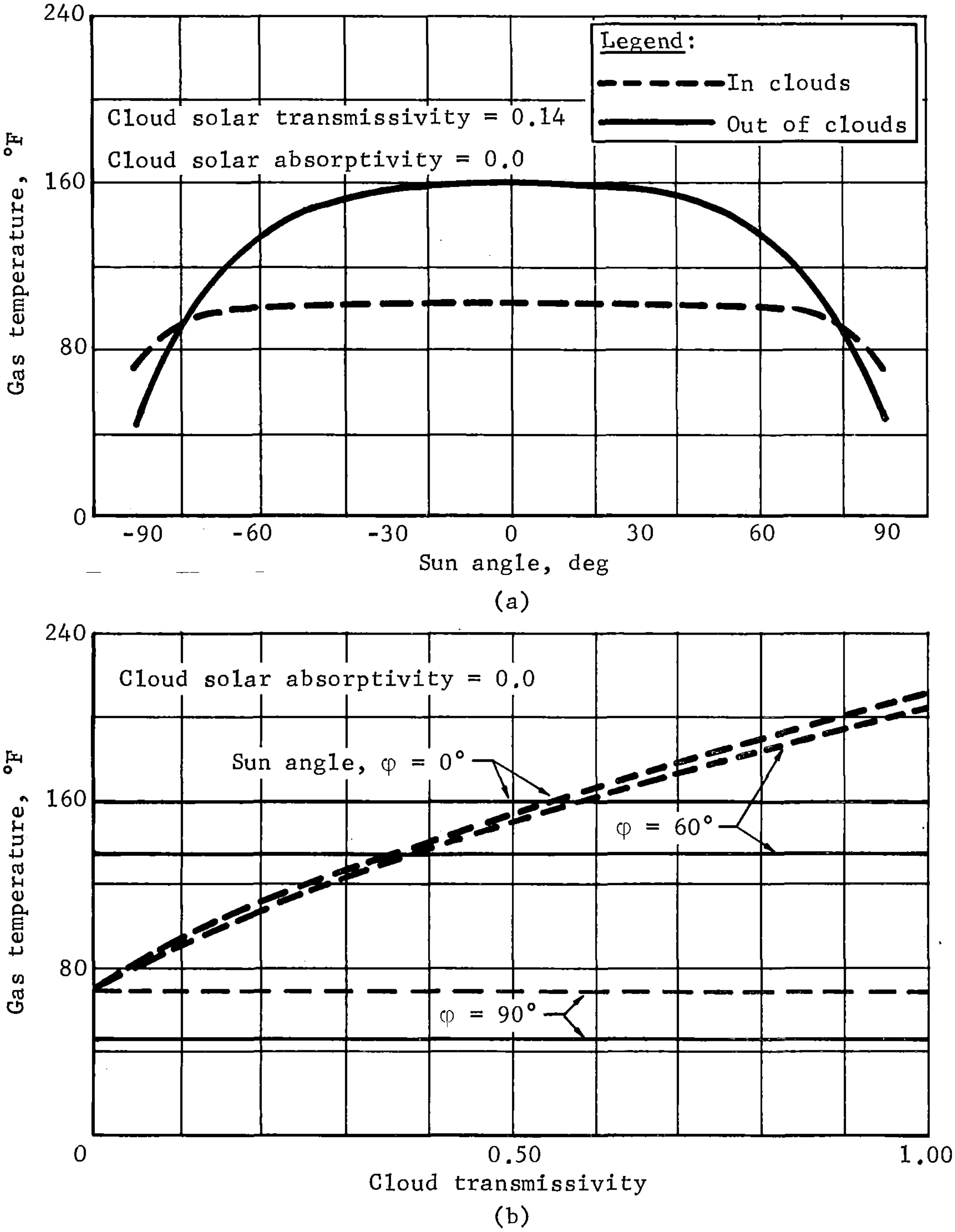
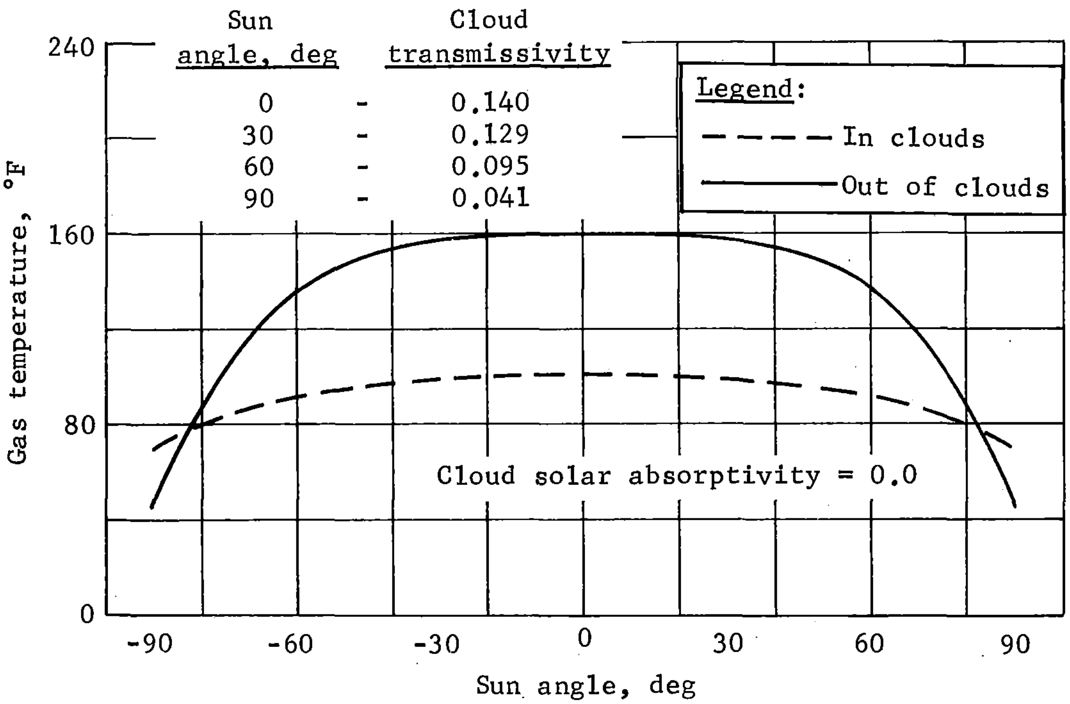
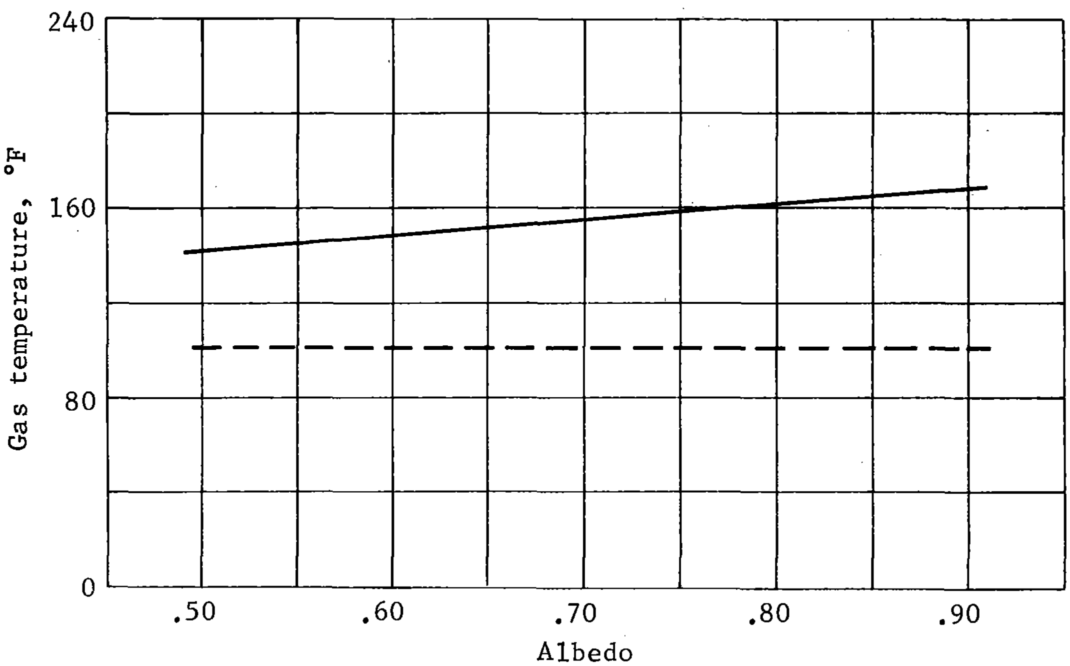


Figure 106.- Gas Temperature Sensitivity to Sun Angle and Cloud Transmissivity



(a)



(b)

Figure 107.- Gas Temperature Sensitivity to Sun Angle and Albedo

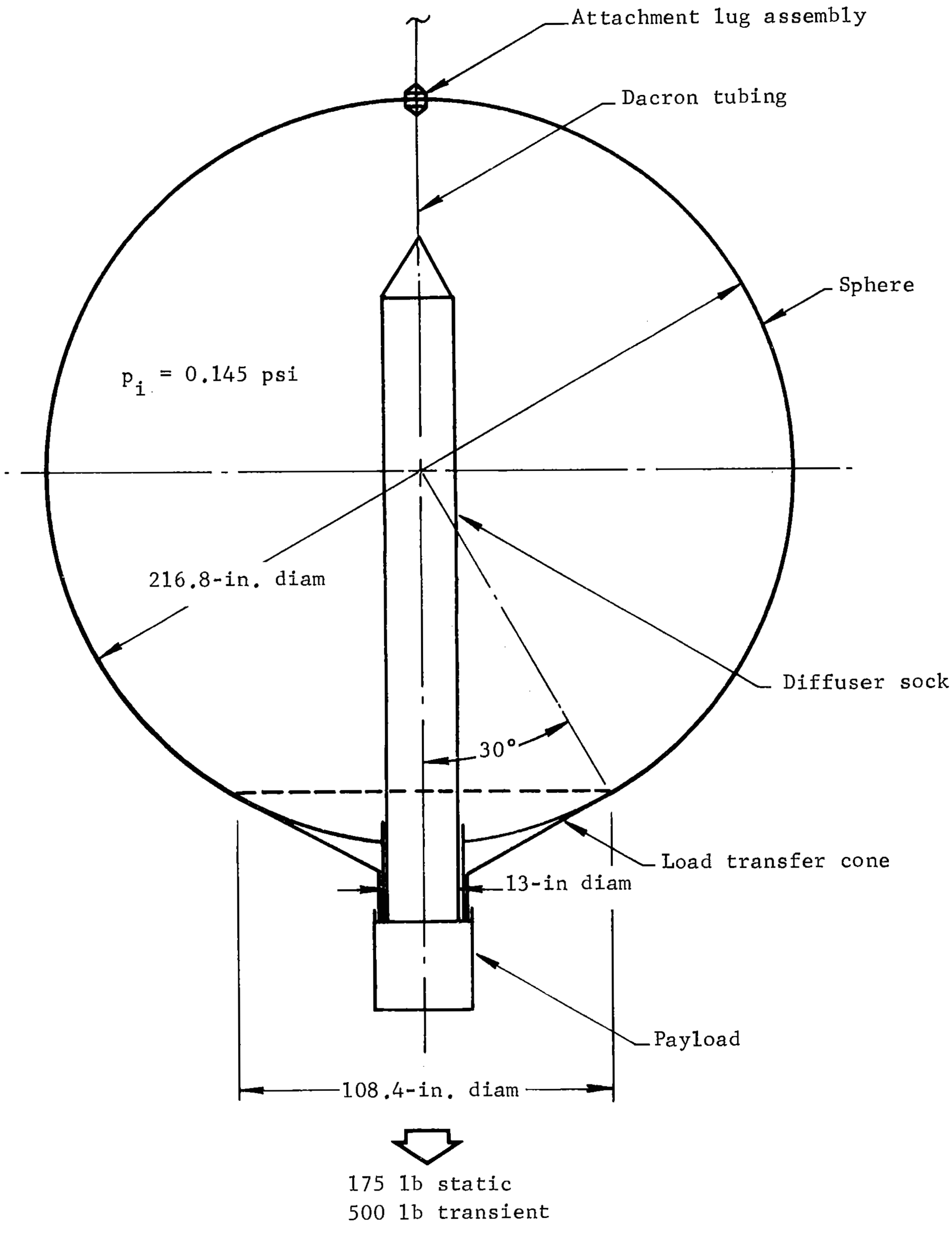


Figure 108.- Balloon General Configuration

Material properties used for this analysis are as follows:

1) Polyester (Mylar) film,
   a) Ultimate tensile strength, F_{tu} = 25 000 psi,
   b) Yield point (at 4%), F_{ty} = 12 000 psi,
   c) Tensile modulus, $E = 0.55 \times 10^6$ psi,
   d) Poissons ratio, $\mu = 0.36$;
2) GT 300 polyester film,
   a) Ultimate tensile strength, F_{tu} = 22 200 psi,
   b) Tensile modulus, $E = 0.55 \times 10^6$ psi;
3) Dacron cloth (1.25 oz/yd^2) - Ultimate tensile strength, P_{ult} = 45 lb/in.;
4) Nylon molding, glass fiber reinforced,
   a) Ultimate tensile strength, F_{tu} = 20 000 psi,
   b) Tensile modulus, $E = 1.4 \times 10^6$ psi,
   c) Ultimate shear strength, F_{su} = 12 000 psi;
5) Dacron webbing, 3/4 in. - Ultimate tensile strength, P_{ult} = 750 lb;
6) Dacron tube - Ultimate tensile strength, P_{ult} = 1000 lb.

Actual membrane stress for the sphere is calculated from wall thickness required to sustain 10 mb (0.145 psig) internal pressure with a membrane stress of 6000 psi

$$t_{req} = \frac{P_i R_p}{2F_o}$$

$$= \frac{0.145 \times 108.4}{2 \times 6000}$$

$$= 0.00131 \text{ in.}$$

Actual wall thickness based on 3/4x3/4x1/2 mil film using film thickness only is:

$$t_{act} = 0.00150 \text{ in.}$$

The actual resulting membrane stress is then:

$$\sigma_{act} = \frac{P_i R_p}{2t_{act}}$$

$$= \frac{0.145 \times 108.4}{2 \times 0.00150}$$

$$= 5250 \text{ psi}$$

Martin of safety:

$$MS = \frac{F_o}{\sigma_{act}} - 1$$

$$= \frac{6000}{5250} - 1$$

$$= +0.14$$

Tube stress: Using the same material for the tube as for the sphere results in:

Hoop stress,

$$\sigma_H = \frac{P_i R_t}{t}$$

$$\sigma_H = \frac{0.145 \times 6.5}{0.00150}$$

$$= 630 \text{ psi}$$

Longitudinal stress,

$$\sigma_L = \frac{P_i R_t}{2t}$$

$$= \frac{0.145 \times 6.5}{2 \times 0.00150}$$

$$= 315 \text{ psi}$$

Margin of safety,

Hoop,

$$MS = \frac{F_o}{\sigma_H} - 1$$

$$= \frac{6000}{630} - 1$$

$$= +8.5$$

Longitudinal,

$$MS = \frac{F_o}{\sigma_L} - 1$$

$$= \frac{6000}{315} - 1$$

$$= +18$$

Unpressurized diameter determination is made by starting with pressurized diameter (see fig. 108)

$$D_{Pi} = 216.8 \text{ in.}$$

at 6000 psi stress. Using 0.0015-in. film at an ultimate pressure of 10 mb (0.145 psig), the induced stress is 5250 psi:

Strain:

$$\epsilon = \frac{\sigma}{E}$$

$$= \frac{5250}{550\ 000}$$

$$= 0.00955 \text{ in./in.}$$

Diameter:

$$D = \frac{D_{Pi}}{(1 + \epsilon)}$$

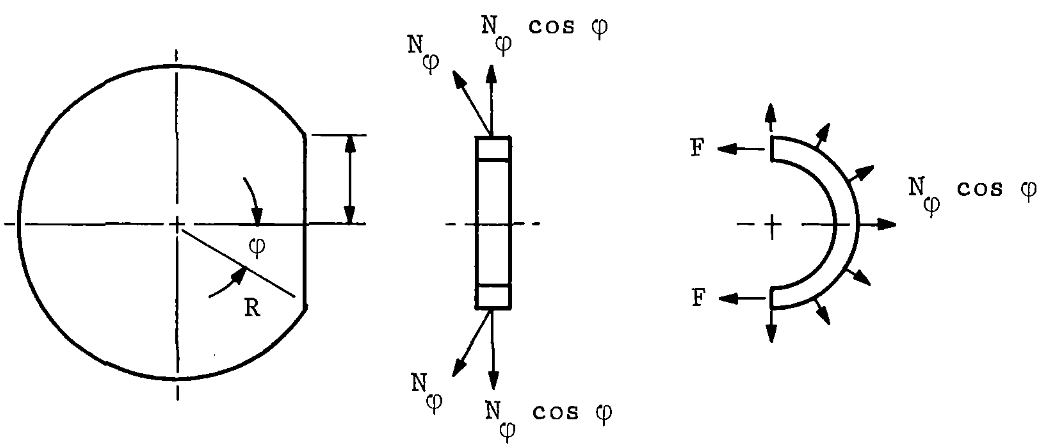
$$= \frac{216.8}{1.00955}$$

$$= 214.75 \text{ in.}$$

or

$$17.95 \text{ ft}$$

Reinforced openings.- The ideal area for reinforcing rings in a spherical shell with a circular segment of the membrane removed may be calculated by equating ring and shell displacement below.



Ring deflection:

$$\delta_r = \frac{\sigma_r \; r}{E_r} \qquad \sigma_r = \frac{F}{A_r}$$

$$\delta_r = \frac{\left(N_\varphi \cos \varphi\right) r^2}{A_r \; E_r} \qquad F = r \; N_\varphi \cos \varphi$$

Shell deflection:

$$\delta_s = r\epsilon_\theta = \frac{r}{E_s\, t}\left(N_\theta - \mu N_\varphi\right) \qquad N_\theta = N_\varphi$$

$$\delta_s = \frac{N_\varphi\, r}{E_s\, t}(1 - \mu)$$

Equating

$$\delta_r = \delta_s$$

and substituting,

$$\frac{\left(N_\varphi \cos \varphi\right) r^2}{A_r\, E_r} = \frac{N_\varphi\, r}{E_s\, t}(1 - \mu)$$

$$A_r = \frac{E_s}{E_r} \cdot \frac{rt \cos \varphi}{1 - \mu}$$

let

$$E_s = E_r,\ \mu = 0.36,^* \ \varphi = 0$$

then

$$A_r = 1.56\ rt$$

With this basic relationship established, it is now possible to determine the amount of reinforcement required at the cut outs.

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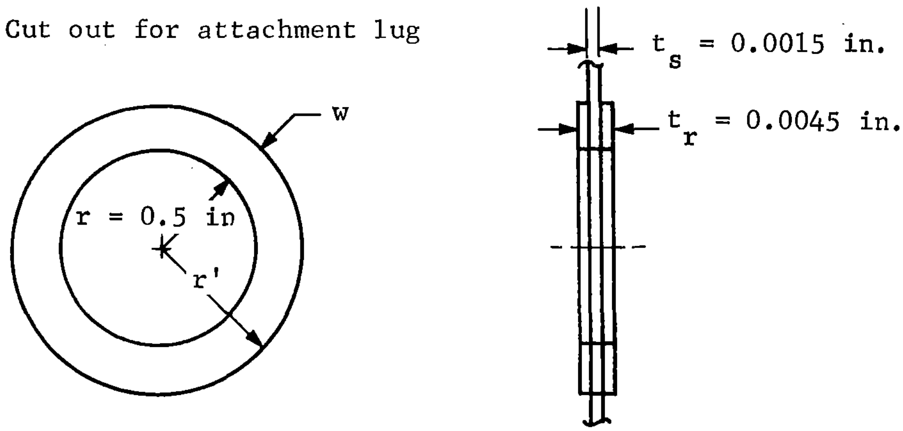
*For polyester resins from reference 11.

Cut out for attachment lug

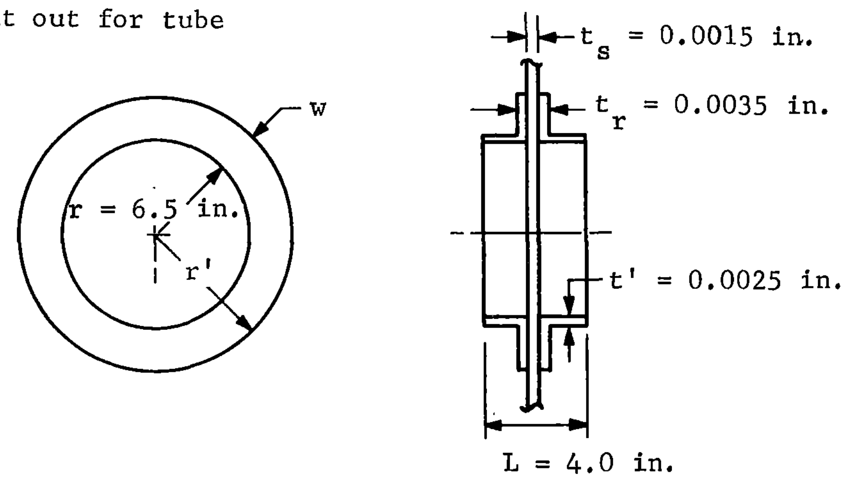


$$A_r = 1.56 \times 0.5 \times 0.0015 = 0.00117 \text{ in.}^2$$

$$w = \frac{A_r}{t_r} = \frac{0.00117}{0.0045} = 0.26 \text{ in.}$$

$$r' = r + w = 0.5 + 0.26 = 0.76 \text{ in.}$$

Cut out for tube



$$A_r = 1.56 \times 6.5 \times 0.0015 = 0.0152 \text{ in.}$$

$$w = \frac{A_r - t'L}{t_r} = \frac{0.0156 - 0.0025 \times 4}{0.0035} = 1.6 \text{ in.}$$

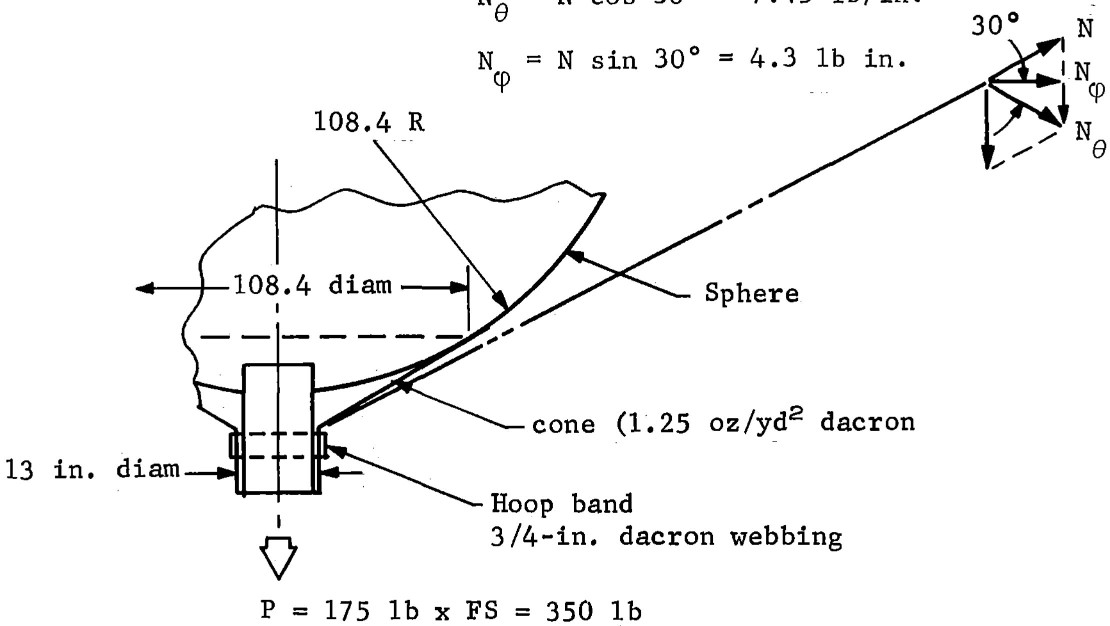
$$r' = r + w = 6.5 + 1.6 = 8.1 \text{ in.}$$

Load transfer cone.-

$$N = \frac{P}{\pi 13} = \frac{350}{\pi 13} = 8.6 \text{ lb/in.}$$

$$N_\theta = N \cos 30° = 7.45 \text{ lb/in.}$$

$$N_\varphi = N \sin 30° = 4.3 \text{ lb in.}$$



Load on hoop band:

$$P_{hoop} = \pi 13\left(N_\theta + N_\varphi\right)$$

$$= \pi 13(7.45 + 4.3)$$

$$= 480 \text{ lb}$$

Margin of safety:

$$MS = \frac{P_{ult}}{P_{hoop}} - 1$$

$$= \frac{750}{480} - 1$$

$$= +0.56$$

Loading due to P enters the balloon structure tangentially through a bonded joint at a diameter of 108.4 in. Stress imposed by this load is then superimposed on the basic membrane stress.

Unit loading due to P:

$$N_P = \frac{P}{\pi d}$$

$$= \frac{350}{\pi 108.4}$$

$$= 1.03 \text{ lb/in.}$$

Unit loading due to P_i:

$$N_{Pi} = \frac{P_i R}{2}$$

$$= \frac{0.145 \times 108.4}{2}$$

$$= 7.85 \text{ lb/in.}$$

Total unit loading:

$$N_t = N_P + N_{Pi}$$

$$= 8.88 \text{ lb/in.}$$

Resulting stress:

$$\sigma_t = \frac{N_t}{t_s}$$

$$= \frac{8.88}{0.0015}$$

$$= 5867 \text{ psi}$$

Margin of safety:

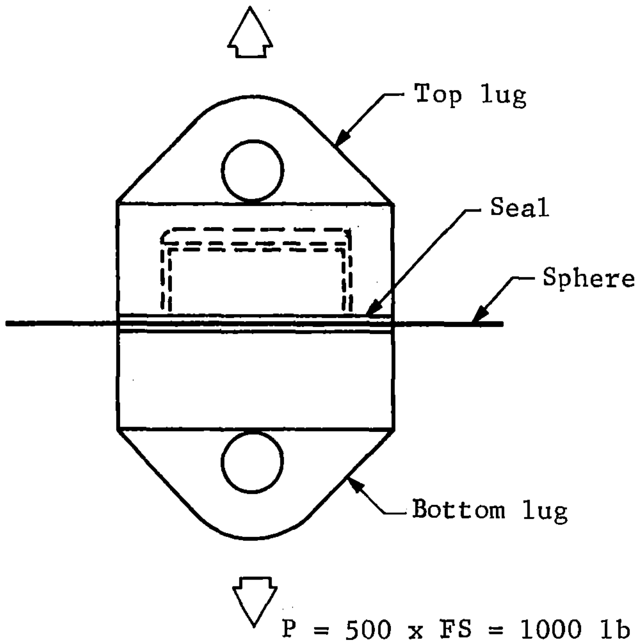
$$MS = \frac{F_o}{\sigma_t} - 1$$

$$= \frac{6000}{5867} - 1$$

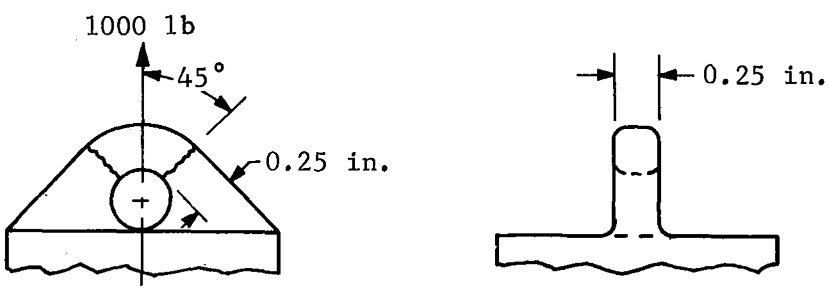
$$= +0.02$$

The cone will be bonded to the sphere with an adhesive not yet determined; however, the loading is of a magnitude such that structural integrity will easily be achieved.

Attachment lug assembly.- The design and loading for the lug assembly is shown below.

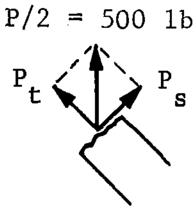


Loading:



Anticipated failure is made at 45° from either side of the applied load with a stress area of:

$$A_s = 0.0625 \text{ in.}^2 \text{ (2 places)}$$



$$P_t = P_s = P/2 \cos 45°$$

$$= 500 \times 0.70711$$

$$= 354 \text{ lb}$$

Shear stress:

$$\sigma_s = \frac{P_s}{A_s}$$

$$= \frac{354}{0.0625}$$

$$= 5650 \text{ psi}$$

Tensile stress:

$$\sigma_t = \frac{P_t}{A_t}$$

$$= \frac{354}{0.0625}$$

$$= 5650 \text{ psi}$$

Margin of safety (shear):

$$MS = \frac{F_{su}}{\sigma_s} - 1$$

$$= \frac{12\ 000}{5650} - 1$$

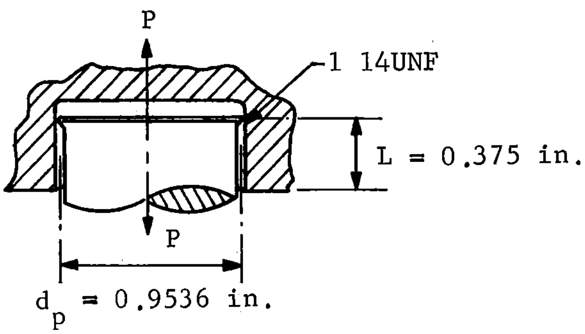
$$= +1.40$$

Margin of safety (tension):

$$MS = \frac{F_{tu}}{\sigma_t} - 1$$

$$= \frac{20\ 000}{5650} - 1$$

Thread shear:



$$\sigma_s = \frac{P}{\pi\ d_p\ 1/2\ L}$$

$$= \frac{1000}{\pi\ .9536 \times .5 \times .375}$$

$$= 1770 \text{ psi}$$

Margin of safety:

$$MS = \frac{F_{su}}{\sigma_s} - 1 = \frac{12\ 000}{1770} - 1$$

$$= +5.8$$

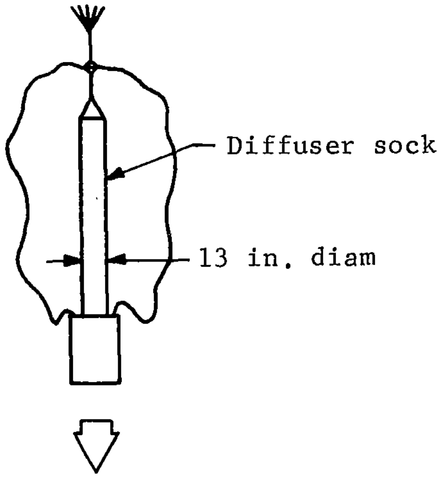
Transient loading through diffuser sock.- Transient loading caused by deployment shock is transmitted from the payload through the diffuser sock to the decelerator system. The tube is constructed of 1.25 oz/yd^2 dacron cloth, which is capable of carrying 45 lb/in. in the load direction.

Unit loading on tube:

$$N = \frac{P}{\pi d}$$

$$= \frac{1000}{\pi\ 13}$$

$$= 24.5 \text{ lb/in.}$$



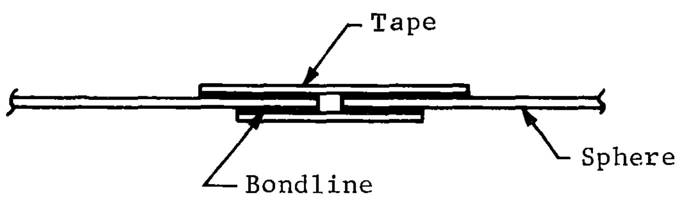
Margin of safety:

$$MS = \frac{P_{ult}}{N} - 1$$

$$= \frac{4.5}{24.5} - 1$$

$$= +0.84$$

Joints.- The jointed areas are constructed by bonding polyester tape on either side of the parent material with a polyester based thermoplastic material. The tape is of the same basic material as the sphere and of greater total thickness. It has been demonstrated through past experience at Raven that this type of construction will develop strength equal to or greater than the parent material. Typical joint construction is shown below.



Transient loading through dacron tubing.- Load is transferred from the diffuser sock to the attachment lug and finally to the decelerating system by dacron tubing that is capable of carrying 1000 lb.

Thus, the margin of safety is

$$MS = \frac{P_{ult}}{P_{act}} - 1 = \frac{1000}{1000} - 1 = 0$$

The problem areas associated with the flotation system operation include:

1) Developing folding and packaging techniques to minimize stresses due to rapid extraction.

2) Minimize stresses due to balloon being subjected to the dynamic pressure of 1 psf. A wind tunnel test program would facilitate evaluation of this problem. Also, the system is designed to initially inflate a bubble at the top of the balloon, which will quickly stabilize the balloon shape. The parachute load will further help to stabilize and stretch the balloon during initial inflation.

3) Leaks due to pinholes must be carefully avoided. Materials and laminate selection as well as packaging and folding techniques will be used to eliminate this problem.

4) The dynamics of initial inflation and recovery was thought to be a problem, but subsequent dynamics simulations of both thermodynamic exchange between the balloon, blowdown inflation gas, and environment; and the trajectory dynamics have proven the system feasibility. It is very advantageous to inflate with cold gas and allow it to warm up, expand and vent off to the required amount for flotation. This allows the system to adapt and initialize itself to variations in temperature and pressure.

5) Another problem area is the unknown magnitude of the variations in the environment such as vertical gusts, cloud breaks, and general turbulence. These perturbations have an impact on the system design and the design sensitivity to these variations is discussed later in this section.

Flotation System Options

Zero-pressure balloon.- There are two possible forms of the nonextensible balloon -- the zero-pressure and the superpressure balloon. It is of interest to examine first why the superpressure balloon was selected for the baseline design and, second, under what conditions the use of a zero-pressure balloon might be considered.

With both forms, a nonextensible balloon when fully inflated fills a virtually constant volume, V_B, because the elasticity of possible balloon materials such as Mylar is negligible. The mass that is supportable by such a balloon is equal to the mass of the displaced atmosphere,

$$M_S = \rho_A V_B \tag{1}$$

where

M_S = total mass of gondola, balloon and inflation gas,

ρ_A = density of atmosphere.

During the original inflation, an extra amount of gas is placed in the balloon so, as the balloon rises to its equilibrium altitude, the balloon is fully extended. During this rise, the superpressure balloon maintains a constant pressure differential, ΔP, above the surrounding atmosphere, while the zero-pressure balloon is vented to the atmosphere so that $\Delta P = 0$.

Equation (1) can be rewritten to separate the mass of the inflation gas from the supported mass:

$$M_L + M_G = \rho_A V_B \tag{2}$$

where

M_G = the mass of gas,

M_L = mass lofted including the gondola or payload mass and the balloon mass.

Assuming that the balloon properly approaches its flotation altitude from below, so that the inflation gas completely fills the balloon, the perfect gas law relates the mass of gas to the fully extended balloon volume, V_B,

$$M_G = \frac{P_G V_B W_G}{R T_G} \tag{3}$$

where

P_G = pressure of gas,

T_G = absolute temperature of gas,

W_G = molecular weight of gas,

R = universal gas constant.

Substituting equation (3) into (2),

$$M_L = V_B \left[\rho_A - \frac{P_G\ W_G}{R\ T_G} \right] \tag{4}$$

The atmospheric density can also be obtained from the perfect gas law,

$$M_L = V_B \left[\frac{P_A\ W_A}{R\ T_A} - \frac{P_G\ W_G}{R\ T_G} \right] \tag{5}$$

In general, the atmospheric and gas pressure are related by the superpressure,

$$P_G = P_A + \Delta P \tag{6}$$

and similarly, the gas temperatures by the supertemperature θ,

$$T_G = T_A + \theta \tag{7}$$

where θ is induced by the balance of radiant heat input and output for the balloon gas. In the operating region on Venus, θ has been determined to be a constant, providing there is a constant cloud cover. For a first-order approximation:

$$T_G = kT_A \tag{8}$$

where $k = 1.059$. Utilizing equations (6), (7), and (8), equation (5) can be written:

$$M_L = M_G \left[\frac{W_A}{W_G} - \frac{1}{k} - \frac{\Delta P}{kP_A} \right] \tag{9}$$

If some perturbation causes loss of gas so that M_G becomes smaller, the superpressure is reduced, in the case of the superpressure balloon, and the balloon remains fully inflated and stable. In the case of the zero-pressure balloon where ΔP in equation (9) is zero, any gas loss requires that this mass lofted, M_L, must be made smaller, or the balloon mass balance is destroyed and the balloon will fall to the planet surface. Zero-pressure balloons can be made stable by dropping ballast, thereby reducing M_L or by providing makeup gas.

Three possible perturbations will probably cause loss of inflation gas -- (1) updrafts that temporarily place the balloon in a region of lower atmospheric pressure; (2) reversable changes in radiant heat flux where the value of θ rises and then returns to a lower level due to the temporary changing of the cloud density; and (3) leakage that is due to the superpressure (in a zero-pressure balloon at the top of the balloon.)

Therefore, as with the baseline configuration, where no provision is made for either ballasting or makeup gas, a zero-pressure balloon cannot survive even the slightest adverse atmospheric changes. Even though the Venus environment can be expected to be more benign than that of Earth where a zero-pressure balloon cannot survive a single sunset, clearly a zero-pressure balloon with no makeup of ballast is undesirable. The primary tradeoff is rather on the proper amount of superpressure. It can be seen from equation (9) that the larger the superpressure for a given balloon, the smaller the payload that can be supported. The tradeoff is then between balloon ability to survive atmospheric perturbations and general payload capability.

It may be desirable to add a gas makeup system to the balloon to provide the balloon the capability of withstanding passing completely out of the cloud cover. If this addition is found necessary, a zero-pressure balloon becomes an attractive alternative design. It is possible to operate a zero-pressure balloon as an altitude sounding device. As previously described, a slight atmospheric disturbance will upset the balloon's mass balance starting the balloon's descent toward the surface. The descent velocity is a function of the magnitude of the disturbance, but can be expected to be on the order of 1 m/sec. When the balloon has descended to a preset temperature level in the atmosphere, makeup gas is injected into the balloon reversing the descent. The balloon will then continue rising to its original flotation altitude to await a second atmospheric disturbance.

With the gas makeup system assumed, the advantages and disadvantages of a zero-pressure balloon are as follows:

1) Disadvantages,

   a) Loses ability to provide wind pattern if balloon cycles between two counter-circulating wind patterns,

   b) Would survive a shorter time than the superpressure balloon if faced with a great number of low amplitude atmospheric variations;

2) Advantages,

   a) Would provide numerous vertical soundings of the atmosphere,

   b) Would survive a longer time than the superpressure balloon if faced with a large number of high amplitude atmospheric variations.

System size and weight.- A study was performed to show the interrelation between float altitude, gondola weight, and balloon size. For this study it was assumed that the balloon had a 1% superpressure, and that the operating balloon skin stress was 3600 psi.

Figure 109 shows the balloon diameter required to float a 175-lb gondola at various radius altitudes from 6100 to 6124 km. The balloon diameter is rapidly increasing at higher altitudes due to both the reduced density and the increasing weight of the balloon itself. The ambient temperatures are 190 and -34°F at radius altitudes of 6100 and 6124 km, respectively. At either of these extremes, the ambient temperatures would result in a requirement for extensive thermal control equipment in the gondola; whereas, at the nominal float radius of 6108 km the ambient temperature is 70°F and active thermal control is not required.

Figure 110 shows the balloon size required for various gondola weights at the nominal radius altitude of 6108 km. The total lofted weight includes the weight of the balloon and inflation gas, which at the nominal gondola weight of 175 lb amounts to an additional 25 lb or a total lofted weight of 200 lb. An approximation of the total BVS system weight including inflation tanks, but not including the entry system weight is:

$$\text{Weight BVS} \cong 2.3 \times \text{weight gondola}$$

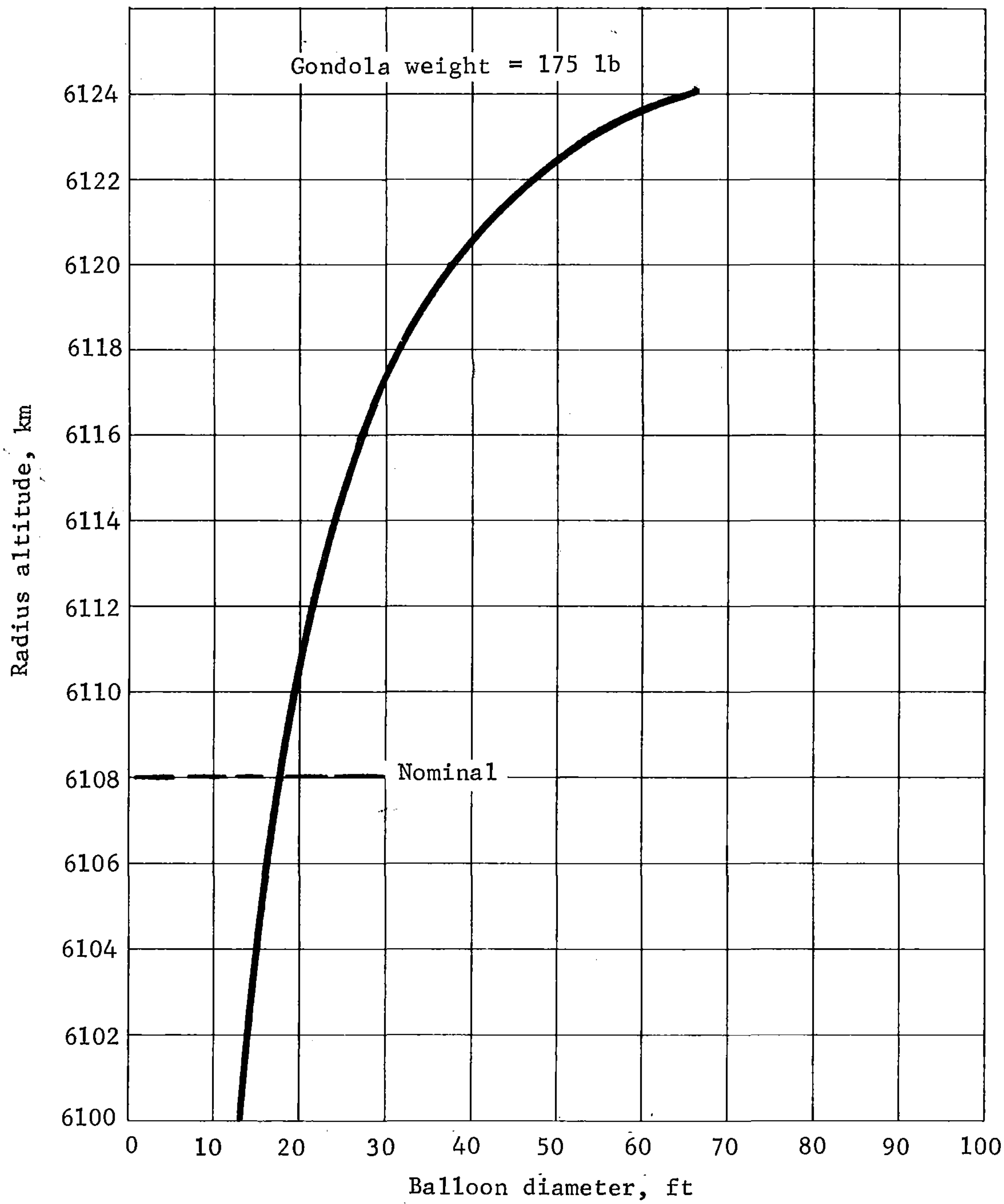


Figure 109.- Float Altitude vs Balloon Diameter

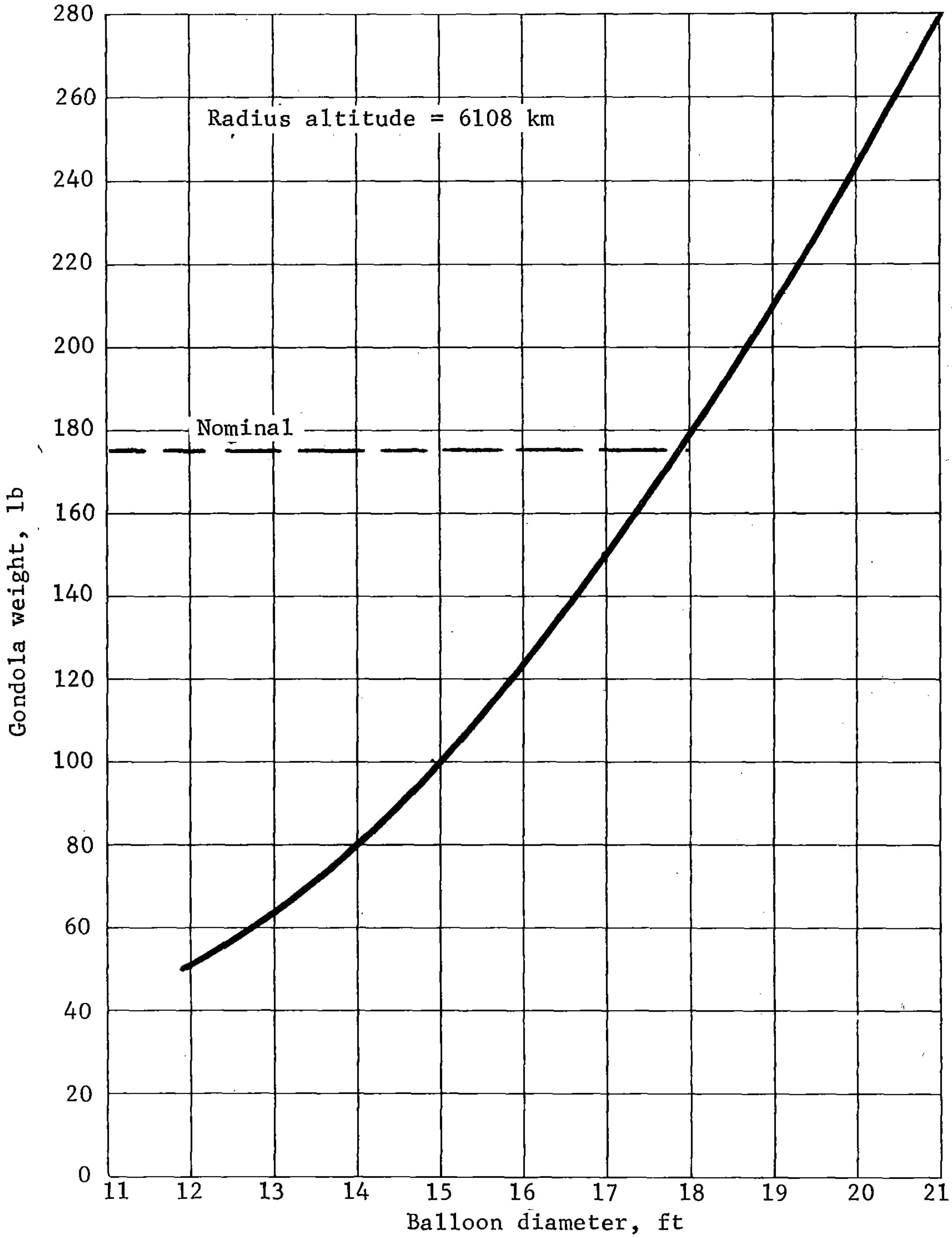


Figure 110.- Gondola Weight vs Balloon Diameter

For the baseline case, a gondola weight of 175 lb results in a BVS weight of 400 lb.

Improved capability.- A study was made to determine the effects of designing the flotation system for improved performance capability. Although the baseline flotation system meets the presently specified requirements, there are some major unknowns that have a direct effect on the probability of mission success. These unknowns concern the general environment and include such items as cloud density and distribution, atmospheric turbulence, and prevailing winds.

When the BVS is disturbed in such a way that it exceeds its present flotation design limitations, it may lose its free lift and descend to the surface, thus terminating the mission. For example, if the balloon is forced to rise due to an updraft beyond about 60 m, it will vent out the superpressure or equivalent free lift and when the updraft stops the balloon descends to the surface. Also, if the BVS floats out of the clouds into the sunlight, it may heat up sufficiently to vent a significant amount of hydrogen. As long as it remains in the sunlight, the balloon will operate normally; however, if it then reenters the clouds, its temperature drops and the balloon will lose its free lift. Crossing the terminator into the dark side results in a similar supertemperature loss. There are three basic ways to correct the situation so that the BVS will continue floating. When the free lift is lost, the balloon may be reinflated with a small makeup gas system or ballast may be released so that the reduced lift will equal the reduced total weight. The third method is actually a change in initial design by constructing a stronger balloon that would carry a very high superpressure and, thus, never exceed its free lift margin during the pressure and temperature variations.

A comparison of these various methods of improving the balloon performance is shown in table 81. Both high-pressure hydrogen tank blowdown and gas generation are shown, as well as the ballast drop and stronger balloon approaches.

For comparison purposes, the case in which the BVS crosses the terminator to the dark side was used. In this situation, the balloon temperature drops about 30°F, and 0.45 lb of makeup gas is required to reinflate the balloon to its nominal superpressure of 6 mb. For Case 1, the high-pressure (4500 psi) H_2 tank has a weight factor of about 14 lb of tank per pound of H_2, and the total weight is about 6.7 lb, including H_2. This system is fairly simple and consists of an on/off valve to release the H_2 and uses the basic balloon pressure sensor switch for control. This system is considered the best overall system at present and the other systems are compared to it.

TABLE 81.- COMPARISON OF SYSTEMS TO IMPROVE BALLOON PERFORMANCE

Type of system	Weight of makeup gas, lb	Total weight of makeup system lb	$\frac{\text{Weight, system}}{\text{weight, } GH_2 \text{ tank}}$, lb	Comments
1. 4500-psi gaseous H_2 tank	0.45	6.7	1.0	Fairly simple
2. Gas generator (lithium borohydride plus water)	0.45	5.3	.8	Most complex; gas must be generated very slowly to keep minimum temperature rise
3. Gas generator (as above) (lithium borohydride plus water) plus a radiator to allow faster gas generation	0.45	11.3	1.7	Most complex as above; balloon gas temperature increase is less than 5°F for a 200 sec total gas generation time
4. Ballast drop (deadweight)	None	8.5	1.3	Simple
5. Ballast drop (two 5-lb drop sondes)	None	10.5	0	No weight penalty since sondes are part of basic BVS
6. Stronger balloon (no makeup gas)	None	52.0	7.8	52 lb is increased balloon weight to handle 35-mb superpressure

The second and third makeup systems include a lithium borohydride and water gas generator. This approach could be the lightest system with sufficient technical development because it requires only 7.24 lb of lithium borohydride plus water for each pound of H_2 gas. It does require a water tank, a canister with catalyst and filter to generate the H_2 gas at 1900°F and to separate the residue and gas as well as an appropriate valve to release the water. Again, the basic balloon pressure sensor switch is used to control the water valve. The gas generator, Case 2, assumes a very slow rate of gas generation so that the generated heat is dissipated without damage to the balloon.

From dynamic simulations it appears that a certain minimum rate may be required for gas generation to keep the balloon from losing excessive altitude. To determine the impact of gas generation rate on the system design, a thermal analysis was conducted to determine radiator weight required. The result is that a radiator tube of 6.0 lb is required to cool the hot gas based on a total generation time of 200 sec and a maximum balloon gas temperature increase of 5°F. Total makeup system weight for Case 3 is 11.3 lb. The gas generation scheme appears slightly heavier and considerably more complex than the high-pressure tank system.

Case 4 is a ballast drop system in which deadweight (lead shot or dense liquid like mercury) is released as required to lighten the BVS so that the reduced lift and weight are again equal. A ballast weight of about 8 lb is required, and a simple valve controlled by the balloon pressure sensor switch could be used. Another possibility, Case 5, would be to reserve the two 5-lb drop sondes until needed for ballast. From a mission standpoint, this is not desirable because the drop sondes would probably be released some distance apart for best coverage. However, with any system, the drop sondes provide additional ballast margin if needed. Because the drop sondes are part of the basic BVS, no weight penalty is shown for them in the comparison.

Case 6 is the approach in which a heavier balloon skin is fabricated giving the balloon the capability of maintaining a superpressure of about 35 mb instead of the basic 6 mb. Because the balloon weight is directly proportional to the internal pressure for a given film stress:

$$W_B = A_B \rho$$

$$t = \frac{pD}{4\sigma}$$

or

$$W_B = A_B \left(\frac{pD}{4\sigma}\right) \rho$$

where

W_B = balloon weight,

A_B = area of balloon,

p = pressure in balloon,

D = diameter,

σ = skin stress,

ρ = balloon film density.

The new balloon weight is about 52 lb and is not competitive from a weight standpoint with any of the other systems. On the other hand, it is completely passive and, therefore, the most reliable approach.

Based on this cursory investigation, a combination system appears desirable. This system would consist of a somewhat stronger balloon in combination with a high-pressure H_2 tank makeup system. In conjunction with the drop sondes, this system would be extremely versatile, and much of the mission might be completed with only the passive more rugged balloon. In the event of excessive turbulence, broken clouds and terminator crossing, the system would have a high probability of surviving the pressure and temperature excursions.

Inflation system stored gas vs gas generator.- Various inflation system approaches have been studied, including a stored gas system and gas generator systems. The stored gas system uses a tank made of a filament-wound thin aluminum liner pressurized with hydrogen to 4500 psi. Boron or carbon filaments promise a weight reduction, but are not considered here. The inflation is performed by opening a valve and letting the tank blow down into the balloon through a diffuser. This results in the hydrogen gas initially entering the balloon at a low temperature (-50°F), and it then slowly heats and expands to inflate the balloon. Ten percent excess H_2 is included and allowed to vent as the balloon pressure tops off at the final float altitude. This allows considerable uncertainty of environmental conditions in which the balloon will adapt and initialize it's temperature and pressure. The blowdown period is nominally 45 sec and is specified to minimize the time during which the balloon is exposèd to the external dynamic pressure while descending on the parachute.

Two types of gas generators were considered -- a hydrazine type, and a metallic hydride (lithium borogydride and water) type. The basic disadvantage of these systems is that considerable heat is generated along with the hydrogen gas. The decomposition temperature of pure hydrazine and hydrazine/water mixtures exceeds 500°F, and the exhaust temperature of the lithium borohydride and water reaction exceeds 1900°F. These extreme gas temperature must be reduced below about 275°F to prevent damage to the balloon materials. More important, when hot gas is used to inflate the balloon, it must eventually cool down to equilibrium temperatures, and it loses pressure and volume in the process. Therefore, a hot gas system does not top off as a cool gas system does. This consideration alone is enough to disqualify the hot gas systems. To make them workable, gaseous and liquid hydrogen were considered for cooling the gas. This results in a doubly complicated dual system for each. For the design comparison, a maximum allowable gas temperature after cooling was chosen at 275°F.

Table 82 compares hydrogen inflation systems on a weight basis. The monopropellant hydrazine gas generators with either liquid or gaseous hydrogen cooling are not competitive with the gaseous hydrogen inflation system by a considerable weight margin, and qualitatively these systems are more complex than the baseline system.

Various chemical reactions, as shown in table 83, were considered for the production of hydrogen gas.

TABLE 82.- COMPARISON OF HYDROGEN INFLATION SYSTEMS

Type of gas generation system	Weight of system / weight of baseline	Gas temperature before cooling, °F	Gas temperature after cooling, °F	Comment
1. Gaseous hydrogen, high-pressure tanks (baseline system)	1.0	-50	----	Baseline system no cooling required
2. Hydrazine gas generator - gaseous H_2 cooling	4.0	500	275	Gaseous H_2 for cooling
3. Hydrazine gas generator - liquid H_2 cooling	2.9	500	275	Liquid H_2 for cooling
4. Metallic hydride plus water gas generator	1.1	1900	275	Gaseous H_2 for cooling
5. Subliming liquids or solids	----	----	----	Not enough vapor pressure at the reference ambient pressure of 612 mb to effectively inflate balloon

TABLE 83.- HYDROGEN CHEMICAL REACTIONS

Compound	Composition	Water reaction	Pound solid and water/lb H_2
Pentaborane	B_5H_9	$B_5H_9 + 15H_2O \rightarrow 5[B(OH)_3] + 12H_2$	13.9
Decaborane	$B_{10}H_{14}$	$B_{10}H_{14} + 30H_2O \rightarrow 10[B(OH)_3] + 22H_2$	15.1
Lithium borohydride	$LiBH_4$	$LiBH_4 + 2H_2O \rightarrow LiBO_2 + 4H_2$	7.25
Hydrazine bis-borane	$N_2H_4(BH_3)_2$	$N_2H_4(BH_3) \rightarrow 5H_2 + 2BN$	6.0

It is apparent that lithium borohydride and hydrazine bis-borane are the most promising from the standpoint of weight of reactants per pound of hydrogen generated. Hydrazine bis-borane is not sterilizable; therefore, lithium borohydride was used in this comparison.

Table 82, Case 4 presents the lithium borohydride comparison and shows that this system is only slightly heavier than the baseline system even with the required cooling gas. However, this system is not developed and is somewhat more complex mechanically; therefore, it has no advantage over the baseline system.

Subliming solids: The subliming solids listed in Table 84 were checked for use as a source of inflation gas. The use of these solids is precluded because of their low vapor pressure and high molecular weight. For example, a system using an inflation gas with a molecular weight of 25 would require a balloon volume of 4550 ft^3 and an inflation gas weight of 284 lb.

The baseline gaseous hydrogen blowdown system was the choice for the BVS inflation system because it is the lightest weight, least complex of those systems considered.

TABLE 84.- SUBLIMING AND DISSOCIATING SOLIDS

Compound	Composition	Mean molecular weight of gas	Vapor pressure at 70°F, psia	Gas
Ammonium bisulfide	NH_4HS	25.5	0.5	$NH_3 + H_2S$
Ammonium carbamate	$NH_4CO_2NH_2$	26.0	0.083	$2NH_3 + CO_2$
Ammonium bicarbonate	NH_4HCO_3	26.3	0.066	$NH_3 + H_2O + CO_2$
Ammonium carbonate	$(NH_4)_2CO_3$	25.8	0.053	$NH_3 + CO_2 + H_2O$
Ammonium azide	$2NH_4N_3$	30	$<10^{-3}$	$2NH_3 + H_2 + 3N_2$
Ammonium formate	$NH_4CO\ OH$	31	$<10^{-4}$	$NH_3 + HCOOH$
Ammonium chloride	$NH_4C\ell$	26.8	$<10^{-7}$	$NH_4C\ell$
Paraformaldehyde	$HO(CH_2O)_nH$	29.3	0.002	$NHCHO + H_2O$
Acetemide	CH_3CONH_2	59	$<10^{-4}$	CH_3CONH_2
Napthalene	$C_{10}H_8$	128	$<10^{-3}$	$C_{10}H_8$
RRC (subliming solid)	----	25	7	----

Flotation System Supporting Data

Probability of survival.- Life expectancy for the Venus balloon is based on data from the GHOST Flight Program, in which some 84 superpressure balloons were launched between March 1966 and March 1967. The launchings were under the auspices of the New Zealand Meteorological Service, the Environmental Science Services Administration, and the National Science Foundation. The flight program was conducted by the National Center for Atmospheric Research and was described in reference 12.

Nineteen of the balloons launched fit into the largest single category applicable to a life prediction for the Venus balloon. The particular balloons were fabricated from 1.5-mil Mylar, with a diameter of 2.04 m, a superpressure of 30 mb, operated in a flotation atmospheric pressure of 300 mb, and were listed as failing because of balloon leakage. It is assumed, in the analysis to follow, that a comparable level of deployment care can be achieved on Venus, and that the integrity of the Mylar film will also be comparable.

The first problem is to develop an empirical distribution function to describe the probability of survival of the subject 19 balloons. Because the form of this distribution is unknown, the two parameter Weibull distribution was used, because the result can be flexibily varied by proper choice of the two parameters,

$$R(t) = e^{-t^{\beta}\alpha} \tag{10}$$

where $R(t)$ is the probability that there will be no balloon life shorter than t, and α and β are distribution parameters.

By taking double logarithms of equation (10), a linear equation is found,

$$\log \log [1/R(t)] = \beta \log t - \log \alpha \tag{11}$$

The actual data for the 19 balloons were then fitted to this linear equation using a least square fit with the following result:

$$R(t) = \exp\left[\frac{-t_E^{0.9296}}{37.25}\right] \tag{12}$$

where the subscript E refers to equivalent earth time.

A second step required to use GHOST balloon data to predict Venus balloon survival probability is the determination of the effect of balloon film thickness on leak rates. During the GHOST program, a set of 30 balloons was subjected to quantitative leak tests before launch. Averaging the leak rates for balloons of two thicknesses and after making adjustments for the varying balloon volumes, two volumetric rates were obtained:

1) 1.5-mil balloon, 0.0004274 $m^3/m^2/day$;

2) 2.5-mil balloon, 0.0001342 $m^3/m^2/day$.

Because these rates were obtained under the same atmospheric conditions, the mass flow per square meter per day will show the same proportion.

To obtain an empirical curve passing through these two points, use was made of two additional points; i.e., zero leak rate with an infinite thickness and an infinite leak rate with zero thickness. Such a constraint produces a relationship with the form:

$$\dot{m}\,\tau^{B} = A \tag{13}$$

where

$\dot{m}$ is the mass leak rate,

τ is the thickness,

A and B are constants.

The ratio of the leak rates for two thicknesses, τ_1 and τ_2 are

$$\frac{\dot{m}_1}{\dot{m}_2} = \left(\frac{\tau_2}{\tau_1}\right)^B \tag{14}$$

The parameter B was determined to be 2.268, and, because it is desired to determine the life expectancy for balloons of film thickness other than 1.5 mils, the mass leak rate for a thickness of τ is given by,

$$\dot{m}_\tau = \dot{m}_{1.5} \left(\frac{1.5}{\tau}\right)^{2.268} \tag{15}$$

Development of balloon leakage model: To model the leakage of gas through the balloon film, the type of flow through possible holes in the film must be defined. A hole large enough to produce a significant leak rate may be considered to be an orifice. For example, a hole of an effective cross sectional area of 1 $(\text{mil})^2$ would act as an orifice because its diameter is in the same order of magnitude as the balloon film thickness. However, 1000 such holes would leak only 5 cu ft of gas per day, initially, for the baseline balloon configuration. Such a collection of holes would still require almost the entire 7-day mission to bleed the 6-mb superpressure down to 1 mb. Not only do smaller holes have a negligible leakage effect, but their existence will be prevented by the use of soft plastic bond between the two laminations of the balloon skin.

Given that the holes under consideration are orifices, the mass flow of gas through the hole is given by,

$$\dot{m} = -YA^* \sqrt{2 \, \Delta P \, \rho} \tag{16}$$

where

A^* = the effective area of the hole including orifice coefficient,

ΔP = the pressure differential across the hole,

ρ = the density of the gas,

Y = expansion factor (varies between 0.995 at 6 mb to 1.0 at 0 mb and assumed to be equal to 1.0).

At this point, we define a random variable C, which is the summation of the area of all holes, each having an effective area of A^*/sq ft of a balloon film with a thickness of 1.5 mils. The mass flow for the entire balloon is then given as follows, after substituting equation (15)

$$\dot{m} = -A \frac{C \sqrt{2 \Delta P \rho}}{\tau^{2.268}} \tag{17}$$

where A is the surface area of balloon. With an appropriate change in C, the surface area of the balloon can be expressed as the ratio of the balloon volume (V) to diameter (D).

$$\dot{m} = \frac{-CV \sqrt{\Delta P \rho}}{D\tau^{2.268}} \tag{18}$$

Because the inflation gas will follow the perfect gas law

$$M = \frac{VM_W \left(P_A + \Delta P\right)}{R \left(T_A + \theta\right)} \tag{19}$$

where

M = mass of gas,

V = volume of balloon,

P_A = atmospheric pressure,

ΔP = pressure across balloon film,

R = universal gas constant,

M_W = molecular weight of inflation gas,

T_A = atmospheric temperatures,

θ = supertemperature of inflation gas.

Differentiating equation (19) with respect to ΔP

$$\frac{dm}{d\,\Delta P} = \frac{V\,M_W}{R\left(T_A + \theta\right)} \tag{20}$$

Substituting the value for dm in equation (20) into equation (18) an equation relating $d\,\Delta P$ and dt is obtained,

$$d\,\Delta P = \frac{-C}{D\tau^{2.268}} \sqrt{\frac{R\left(T_A + \theta\right)}{M_W}} \sqrt{\Delta P\,\left(P_A + \Delta P\right)}\,dt \tag{21}$$

Equation (21), after integrating both sides between the limits of the original superpressure, ΔP_O at time $t = 0$ to a decay limit of 1 mb at time t results in the following relation,

$$\log\left[\frac{\sqrt{P_A + 1} + 1 + P_A \quad 2}{\sqrt{\Delta P_O\,\left(P_A + \Delta P_O\right)} + \Delta P_O + P_A/2}\right] = \frac{-C\,t}{D\tau^{2.268}} \sqrt{\frac{R\,\left(T_A + \theta\right)}{M_W}} \tag{22}$$

The product Ct equals a constant for any given set of atmospheric and balloon parameters. To relate the Venus balloon to the known experience of the Earth-launched GHOST balloons, the product Ct can be evaluated for the same conditions the empirical life expectancy relationship equation (12) was developed, where,

$P_A = 300$ mb $\quad \Delta P_O = 30$ mb $\quad D = 6.63$ ft $\quad T_A = 223.6°K$

$\theta = 0°$ $\quad M_W = 4.032$ $\quad \tau = 1.5$ mils

For this set of conditions,

$$Ct_E = 0.030915 \tag{23}$$

If the balloon film in the Venus balloon is manufactured, handled, and deployed in a similar manner to that used in the GHOST program, the random variable C in equation (23) had the same distribution it will have applied in equation (22) for Venus usage. Therefore, substituting equation (23) into equation (22) and solving for t_E,

$$t_E = \frac{\dfrac{0.030915\ t \sqrt{R\left(T_A + \theta\right)/M_W}}{D_T^{2.268}}}{\log\left|\dfrac{\sqrt{\Delta P_O\left(P_A + \Delta P_O\right)} + \Delta P_O + P_A/2}{\sqrt{P_A + 1} + 1 + P_A/2}\right|} \tag{24}$$

The equivalent earth time required for the calculation of the reliability or survival probability in equation (12) is found by means of equation (24) as a function of the desired Venus operating time and the appropriate atmospheric and Venus balloon parameters.

Figure 111 plots the survival probability of two balloon inflation gases -- hydrogen and helium -- as a function of the total BVS weight required to support a 175-lb gondola for two mission times of 1 and 7 days. The dashed portion of the curves represent expenditures of so little weight for the balloon film that the stress exceeds the specified 3600 psi. It is apparent, from figure 111, that hydrogen as an inflation gas is clearly superior to helium, even when the reliability effects of increased film stress are ignored. It is impossible to assess the importance of this reduced stress until material testing provides some indication or the variance of the tensile strength of Mylar after sterilization and at various temperatures. As an indication of this effect, the stress developed in a 420-lb BVS with hydrogen is 1850 psi while a helium balloon for the same weight station is 3300 psi. Because the yield stress of Mylar after sterilization is in excess of 12 000 psi, the reliability difference in these two levels will probably be negligible.

Balloon materials and fabrication.- The requirements for the balloon materials and fabrication techniques imposed by the BVS design criteria led to a broad review of the state of the art in balloon technology. Most balloon experience has been gained with the large zero-pressure stratospheric balloon systems. Recently, however, an extensive program has been undertaken by the National Center for Atmospheric Research (NCAR). This program, GHOST (global horizontal sounding technique), uses free-floating superpressure balloon. In 1966 and 1967, 100 balloons were launched and considerable success was experienced. Average lifetime at 200 mb altitude was three months with one balloon staying afloat over 365 days.

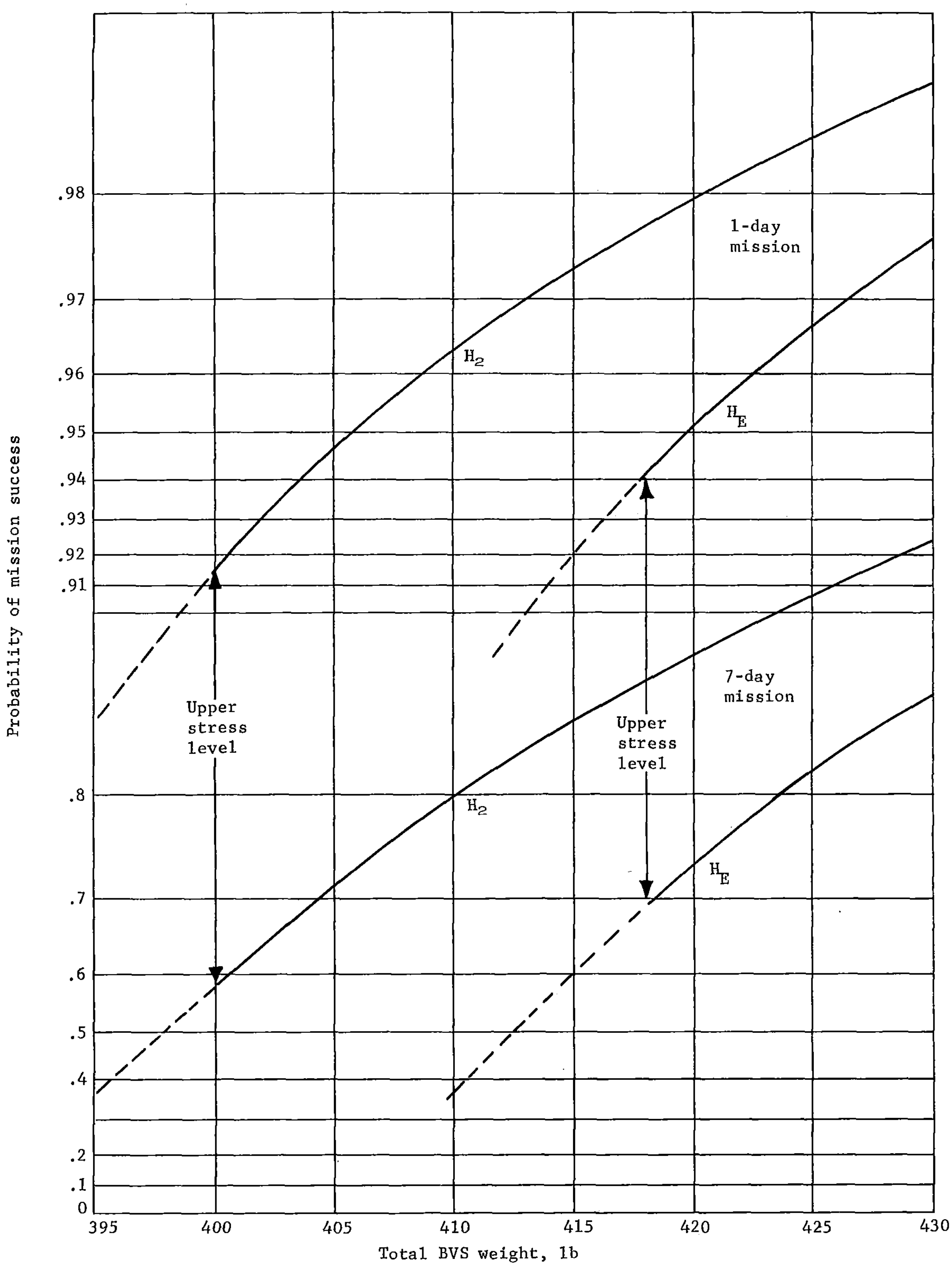


Figure 111.- Reliability Comparison, Hydrogen vs Helium Inflation Gas

Therefore, there is considerable evidence that a superpressure balloon could be constructed and flown on the BVS mission with a high degree of mission probability of success.

Raven Industries, Inc., made a materials survey and evaluated materials against the balloon requirements. The selection of all materials used in the balloon assembly was based on the following factors:

1) Strength to weight ratio;

2) Permeability characteristics;

3) Resistance to sterilization;

4) Handling characteristics and durability;

5) Sealing and assembly characteristics.

The evaluation of candidate materials used inputs derived in a Raven Industries, Inc., superpressure balloon packaging technique study funded by NCAR under Contract 63-68.

The material selected is a specialized assembly of polyester film layers. This material is composed of a trilamination of 3/4-mil film, 1/2 mil adhesive, 3/4 mil film. This composite has the strength characteristics of 1 1/2-mil polyester film with a unit weight comparable to 2-mil polyester film, that is, 0.0149 lb/sq ft. Although the adhesive layer adds significantly to the unit weight, added resistance to sterilization will be realized with a thick adhesive layer according to the Arvey Corporation. The desirability of this resistance is apparent in view of sterilization tests conducted by the Martin Marietta Corporation on a 3/4x3/4-mil bilamination of polyester film.

Research being conducted under NCAR Contract 63-68 has determined some techniques of minimizing film damage during fabrication, shipping, and packing. The most promising of the techniques analyzed to date is a layer of 1/8-in. thick, strippable, polyurethane foam.

The foam is banded to the film surface with a latex adhesive that remains soft and is easily stripped off. Use of such a material provides a cushion for the film during those phases of the fabrication process during which the material may be subjected to motion across abrasive surfaces. During fabrication, enough of the gore edge may be exposed for sealing. The coating may remain on the balloon until the actual packing operation is initiated. At this time, the coating can be stripped off as the balloon is folded, leaving those surfaces in contact with other surfaces coated until the balloon is inserted in the container.

The most serious problem in the generation of pinhole leaks during handling and packing a balloon is the three-cornered fold or crease. Any abrasion on the point of such a crease will result in a pinhole formation. It is not possible to avoid the formation of this type of fold in packing the balloon into the configuration required. Therefore, preparation of the container to minimize potential abrasion must be accomplished.

To deploy the balloon in the most orderly fashion, the basic folding technique must result in an accordian-folded system capable of being deployed by unfolding from the top. The pack will be folded in the following manner:

1) Fold balloon lengthwise gore-on-gore to form a system one gore wide;

2) Fold one edge over the upper surface of the gore-on-gore fold, and fold one edge under the lower surface to form an S-shaped cross section. The width of the balloon will be compatible with the container diameter at this point;

3) Accordian-fold the balloon into the container. To minimize abrasion, special techniques such as lowering the folded balloon vertically into the container may be required.

Completion of the evaluation of balloon packaging under the previously referenced contract will more fully define potential pinhole prevention techniques. Although the general pinhole sensitivity of materials has been studied, no significant work has been accomplished in the determination of the effect of a dynamic deployment on this factor.

A survey of potential balloon materials (ref. 13) has revealed that the choice of materials is limited. Because of its generally favorable properties and the accumulation of flight experience, Mylar polyester film (polyethylene terephthalate) is a logical starting point in a materials investigation (ref. 14).

The tensile stress-strain curve for Mylar (fig. 112) displays a yield point at approximately 13 000 psi (room temperature) and 7% elongation. Thereafter, the film continues to deform with ultimate failure occurring at an elongation of 125% and a stress of 23 000 psi. The second-order transition temperature of Mylar is reported to be 80°C. The effects of temperature on physical and mechanical properties is relatively small between -20 and 80°C, and no embrittlement occurs at temperatures as low as -60°C. Above 80°C, more marked changes in tensile properties are noted (fig. 113), although Mylar retains useful properties up to 150 to 175°C.

Mylar will shrink when exposed to elevated temperature. At 150°C in air, this shrinkage amounts to approximately 2%. Practically all of this change occurs during the first 10 minutes of exposure to elevated temperature and the film, once shrunken, retains dimensional stability during prolonged exposure at or below the particular temperature. This suggests that Mylar film used in balloons subjected to sterilization must be preshrunk before laminating and fabrication.

Also of concern are the possible changes in crystallinity or crosslinking density of Mylar during prolonged exposure to elevated temperature (sterilization), with the resulting changes in mechanical properties and permeability. Data reported by DuPont (figs. 114 and 115) on the effects of heat-aging Mylar show that an exposure of 100 hr at 150°C in air causes the tabulated changes as shown below in room temperature properties.

Property	Before exposure	After exposure
Tensile strength, psi	23 000	22 000
Tensile modulus, psi	530 000	510 000
Elongation, %	127	113
Tear strength, g/mil	23	12

Thus, while tensile strength, modulus, and elongation are virtually unchanged, propagating tear strength appears to be significantly affected.

Kapton polyimide film has properties very similar to those of Mylar (fig. 116) except that its thermal stability is significantly higher. Thus, if Mylar is found to be seriously degraded as a result of sterilization, Kapton becomes the next logical choice of balloon material. Kapton shrinkage is only 0.3% at 250°C. This shrinkage is caused by residual stresses placed in the film during manufacture. The initial temperature exposure relieves the residual stresses and no further shrinkage will occur thereafter. Aging for up to 1000 hr at 300°C in helium has no effect on the tensile strength or elongation of Kapton film (figs. 117 and 118).

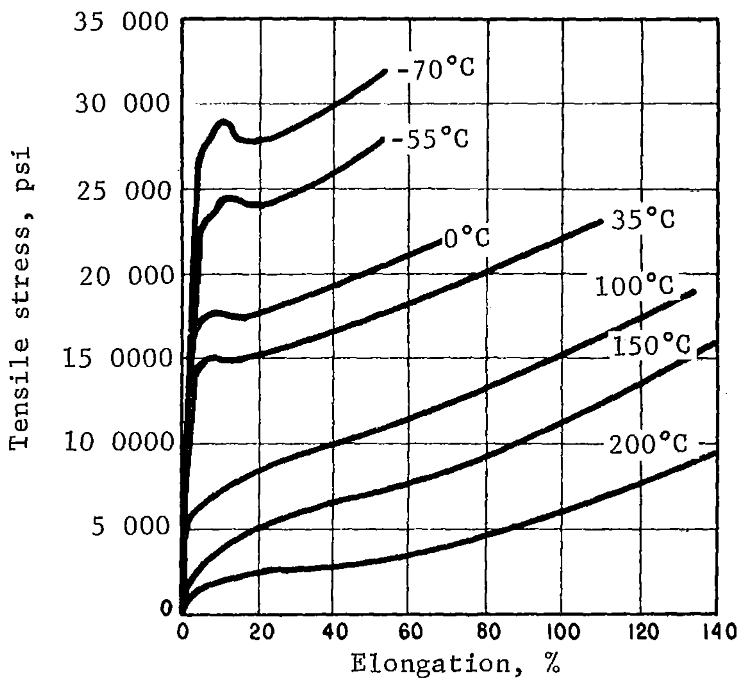


Figure 112.- Stress/Strain Curve for Mylar

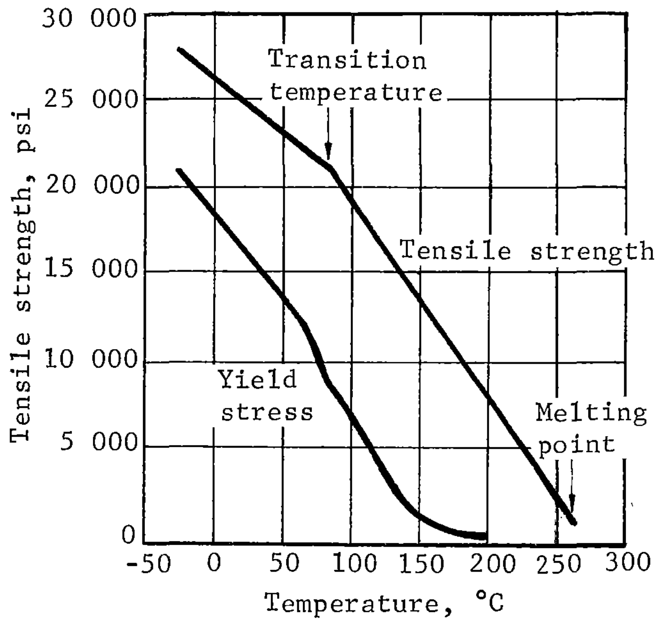


Figure 113.- Tensile Strength vs Temperature for Mylar

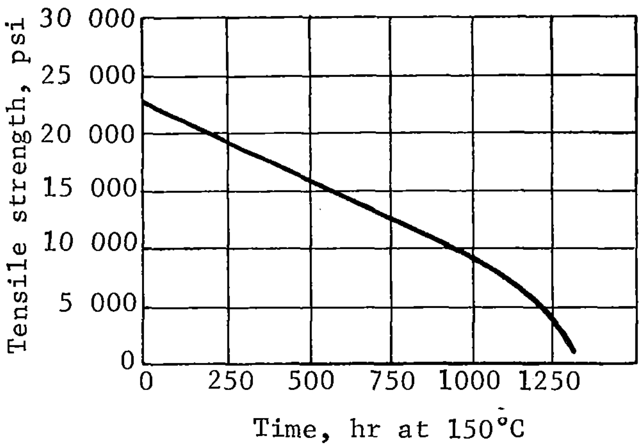


Figure 114.- Tensile Strength of Mylar after Heating in Air

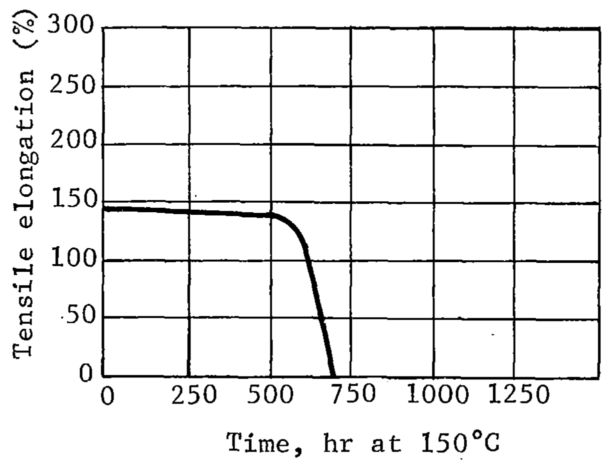


Figure 115.- Tensile Elongation of Mylar after Heating in Air

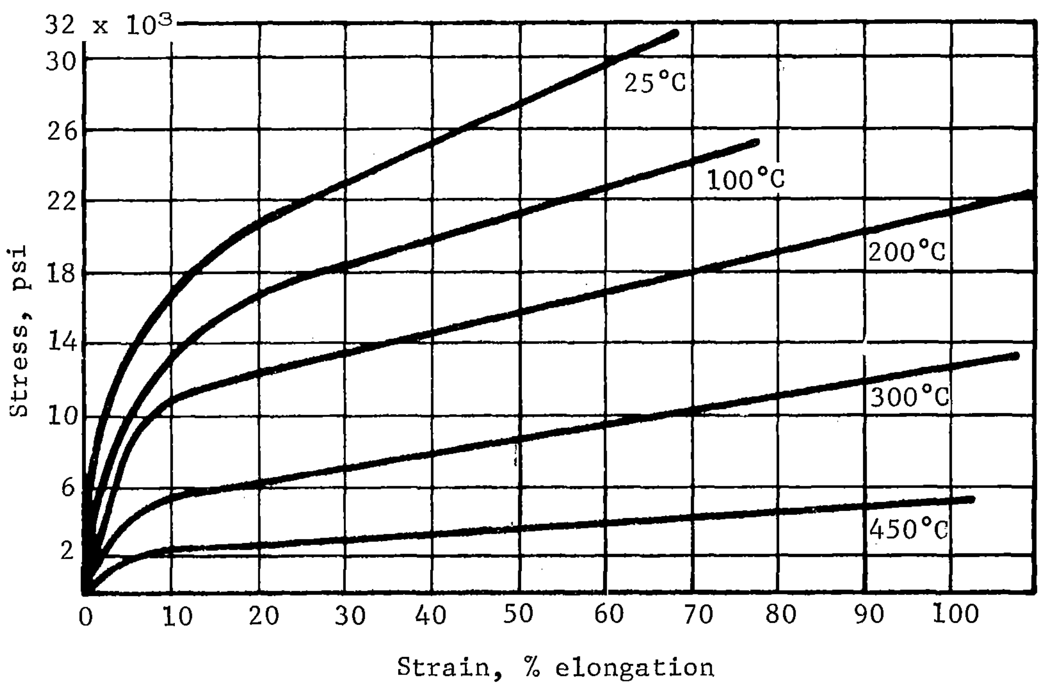


Figure 116.- Tensile Stress/Strain Curves for 1-mil Kapton

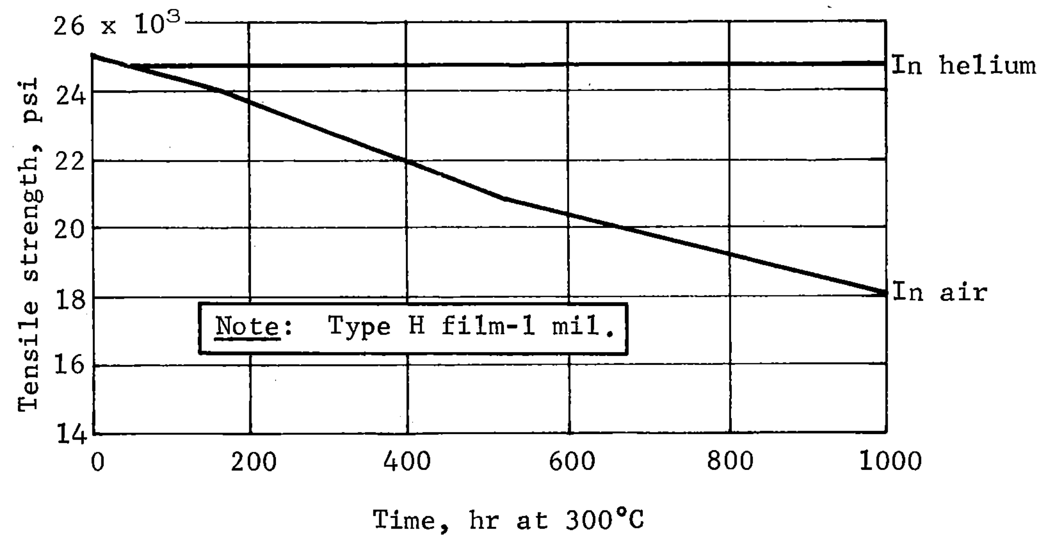


Figure 117.- Tensile Strength vs Aging at 300°C

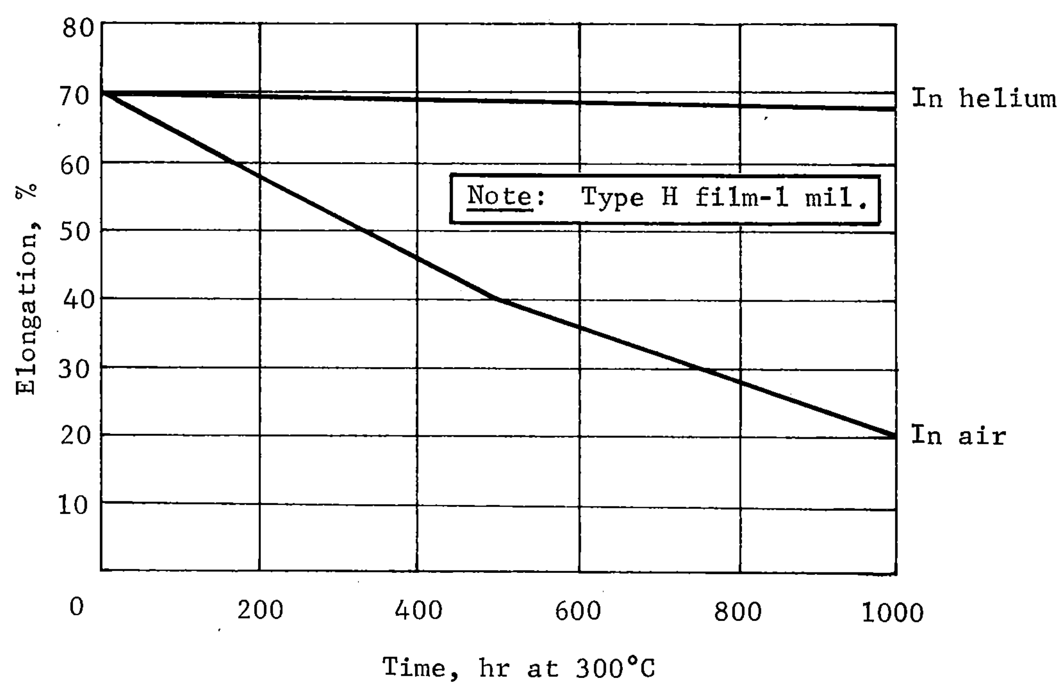


Figure 118.- Ultimate Elongation vs Aging at 300°C

With the exception of better thermal stability, Kapton offers no particular advantage over Mylar, as shown by the comparison of room temperature properties in table 85. In fact, on the basis of its greater elongation, folding endurance, and tear strength and its lower permeability to hydrogen, Mylar is the preferred film material.

TABLE 85.- COMPARISON OF ROOM TEMPERATURE PROPERTIES OF MYLAR AND KAPTON

Property	Mylar	Kapton
Ultimate tensile strength, psi	23 000	25 000
Yield point, psi	12 000	10 000
Ultimate elongation, %	120	70
Stress to produce 5% elongation, psi	13 000	13 000
Tensile modulus, psi	550 000	430 000
Impact strength, kg-cm/mil	6	6
Folding endurance, cycles	> 100 000	10 000
Tear strength, propagating, g/mil	15	8
Tear strength, initial, g/mil	900	510
Density, g/cm^3	1.39	1.42
Moisture uptake at 50% RH, %	.2	1.3
H_2 permeability, cm^3/cm^2-sec-cmHg/mil	0.236×10^{-7}	0.590×10^{-7}

Effect of inflation tank fabrication on gondola weight.- The gondola weight supportable with a given overall system weight is a strong function of the weight of the inflation gas tanks. The weight of the displaced atmosphere can be equated to the difference between the total system weight and the weight discarded, which is primarily the tank weight. This difference is also equal to the weight lofted by the balloon,

$$V_B \rho_A = W_{total} - W_{discarded} = W_{lofted} \tag{25}$$

where

V_B = balloon volume,

ρ_A = density of atmosphere (0.06588 lb/ft^3 at 6108 km).

It is convenient to express the tank weight in terms of tank weight per pound or hydrogen, which is approximately a constant even though the tanks are not spherical. Using a total system weight of 400 lb, support equipment weight discarded of 58.5 lb and a 10% excess gas allowance, equation (25) is rewritten

$$V_B \rho_A = 341.5 - \text{gas discarded} - \text{tanks discarded}$$

$$= 341.5 - 0.1\, V_B \rho_G - 1.1\, K\, V_B \rho_G \tag{26}$$

where ρ_G is density of the inflation gas, which is 0.003004 lb/ ft^3 at baseline conditions.

Equation (26) can be rearranged to provide an expression for the required balloon volume as a function of the tank weight to gas weight ratio, K,

$$V_B = \frac{341.5}{0.06618 + 0.003304\, K} \tag{27}$$

For a given balloon film stress level, balloon film density, and superpressure the balloon film weight per cubic root of volume is a constant, ρ_B,

$$\rho_B = 0.004806\ lb/ft^3 \tag{28}$$

Returning to equation (27), the atmospheric weight displaced must equal the weight lofted and can be broken down as follows:

$$V_B \rho_A = W_{gondola} + W_{balloon} + W_{gas} + W_{balloon\ fittings}$$

$$= W_{gondola} + V_B \rho_B + V_B \rho_G + 1.75 \tag{29}$$

Solving for the gondola weight,

$$W_{gondola} = V_B \quad \rho_A - \rho_B - \rho_G \quad - 1.75 \tag{30}$$

Substituting equation (27) for balloon volume,

$$W_{gondola} = \frac{341.5\ (0.05807)}{0.06618 + 0.003304\ K} - 1.75$$

$$= \frac{19.83}{0.06618 + 0.003304\ K} - 1.75 \tag{31}$$

Figure 119 plots the gondola weight that can be supported as a function of the tank weight to gas weight ratio K. The following conditions are applicable:

Total system weight	400 lb
Skin thickness (total)	1.75 mils
Balloon film density	98.3 lb/ft^3
Balloon fittings weight	1.75 lb
Balloon stress (carried by 1.3 mils) . .	3600 $lb/in.^2$
Support equipment discarded (parachute tank support truss and inflation controls)	58.5 lb
Balloon superpressure	6 mb
Balloon supertemperature	31°F
Atmospheric density	0.06588 lb/ft^3

Indicated in figure 119 are three valves for K corresponding to three types of tank fabrication:

Fabrication type	K
Glass wrap with aluminum liner	14 lb/lb
Titanium with proper liner	19.46 lb/lb
Steel	40 lb/lb

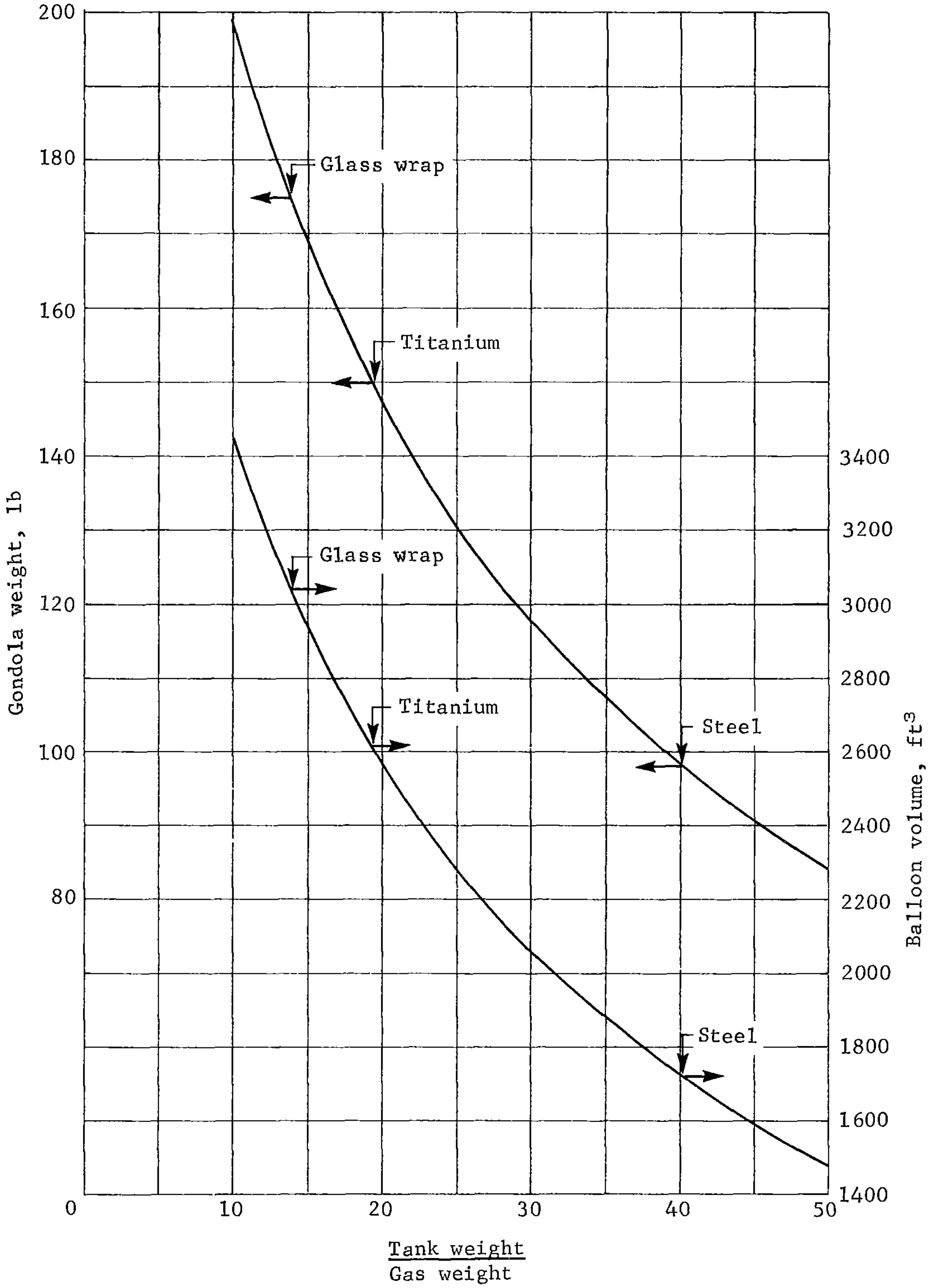


Figure 119.- Gondola Weight as a Function of Tank Weight to Gas Weight Ratio

CAPSULE PROPULSION

Summary.- The capsule propulsion requirements for the orbital, flyby, and Venus/Mercury swingby missions allow the use of space-proved engines and components. The mission requirements differ only in magnitude of impulse and the desire for a variable impulse for the deorbit mode. A summary of the propulsion requirements are shown in Table 86.

Deorbit propulsion subsystem.- The propulsion subsystem shown in figure 120 is a pressure-fed bipropellant system using a single Lunar Orbiter engine to provide the required deorbit thrust. The system is designed to provide a velocity increment from 100 to 250 m/sec to an 840-lb capsule. The propellants used are N_2O_4 and 50-50 UDMH/hydrazine.

The subsystem is pressurized using regulated gaseous nitrogen. Squib-operated valves isolate the pressurant and propellants before system activation. Engine ignition and shutdown are controlled by solenoid-actuated valves mounted on the engine. On completion of the burn, the oxidizer and fuel ullage volumes are isolated by squib-actuated valves. The overall propulsion system characteristics are shown in table 87, the system weight statement is shown in table 88.

Thrust vector control is accomplished by spin stabilization of the capsule. Two oxidizer, two fuel, and two pressurant tanks are located symmetrically about the capsule spin axis.

TABLE 86.- CAPSULE PROPULSION REQUIREMENTS

Requirement	Orbital mission	Flyby mission	Venus-Mercury mission
Velocity increment, m/sec	250 maximum	50 ± 0.5	93 ± 0.5
Capsule weight, lb	840	960	1045
Total impulse, lb_f-sec	23 700	4 980	10 550
Fixed/variable impulse, m/sec	Variable, 100 to 250	Fixed	Fixed
Use developed engine, thrust, lb_f	Lunar Orbiter 100	Mariner '69 50	Mariner '69 50
Sterilizable	Yes	Yes	Yes
Spin during thrusting, rad/sec	2.2 rad/sec	2.2 rad/sec	None
Thrust misalignment	.5° along thrust axes	.5° along thrust axes	.5° along thrust axes.

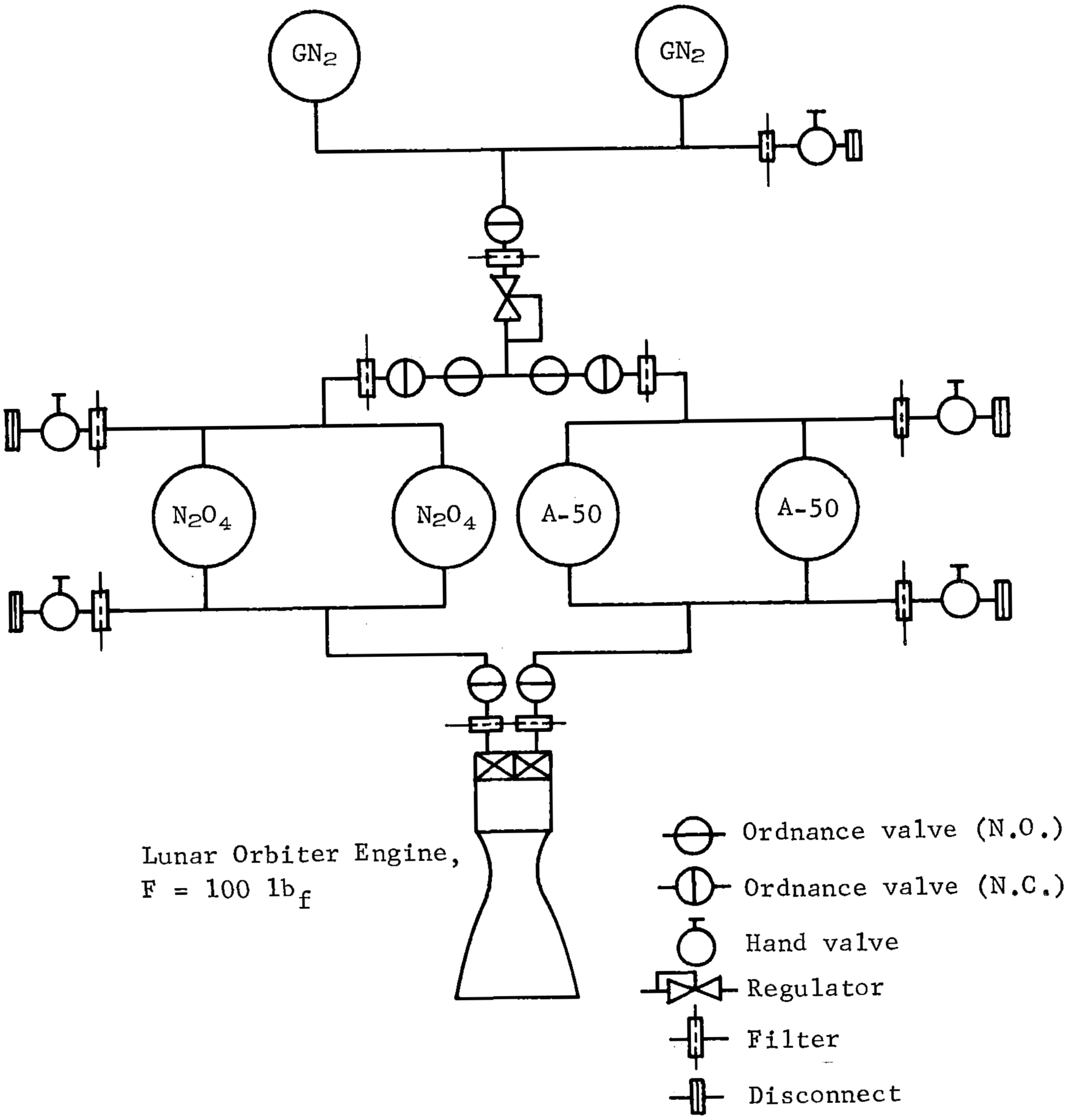


Figure 120.- Deorbit Propulsion Subsystem

TABLE 87.- DEORBIT PROPULSION SUBSYSTEM CHARACTERISTICS

Engine thrust, lb	100
Chamber pressure, psia	100
Specific impulse (delivered), sec	295
Area ratio	50:1
Mixture ratio (O/F)	1.6:1
Maximum burn time, sec	237
Fuel tank volume, cu ft	0.33
Fuel tank diameter, in.	10.3
Oxidizer tank volume, cu ft	0.37
Oxidizer tank diameter, in.	10.8
Pressurant tank volume, cu ft	0.04
Pressurant tank diameter, in.	5.3
Operating temperature range, °F	40 - 80
Tank pressure, psia	200
Initial pressurant tank pressure, psia	3600

TABLE 88.- DEORBIT PROPULSION SYSTEM WEIGHT STATEMENT

Item	Weight, lb	
Total propellant		84.41
Usable fuel	30.92	
Usable oxidizer	49.49	
Total usable	80.40	
Total trapped	4.01	
Fuel tanks (2)		2.84
Oxidizer tanks (2)		4.45
Pressurant		1.67
Pressurant tanks (2)		2.18
Engine		5.50
Components and lines		16.80
Engine mounts, supports		13.56
Total wet system weight		131.44
Total dry system weight (including nonusable propellant and gas)		51.04

To verify the selection of the Lunar Orbiter engine for the deorbit propulsion system, an analysis was performed to compare the preferred design with solid motors and the Mariner '69 engine system.

Figure 121 presents propulsion system weight data as a function of total impulse. This analysis indicates that weight can be saved by using a solid motor for deorbit. This weight advantage is offset by comparing the development risk of the three systems. The Lunar Orbiter engine has been fired in space on five flights and has demonstrated a high reliability. The engine, including valves, was sterilized by Martin Marietta Corporation under JPL Contract 951709, and no deleterious effects were incurred as a result of sterilization.

The Mariner '69 engine will have flown in space before the BVS launch. Sterilization of this engine does not represent a development problem in that the engine does not have valves. However, this system was eliminated from consideration due to its greater weight, as shown in figure 121.

The solid motor represents a development risk in that the effects of sterilization on an entire motor are not known. However, the solid motor represents a strong alternative for the deorbit system due to its weight advantage and simplicity.

The major development problem for the solid motors is demonstration of sterilization compatibility. Small samples of propellant have successfully been sterilized, however, no flight motor configurations have been sterilized. Therefore, the effects of sterilization on igniter performance, case bonding, insulation, nozzle design, and other critical areas have not been determined.

A solid motor requires poststerilization checkout in the critical areas of igniter, propellant grain, case bonding, and nozzle. Electrical continuity checks can be performed on the igniter to ensure that it is flight-ready after the terminal sterilization cycle. Nondestructive test methods must be used to check the remaining critical areas. The potential candidates for nondestructive testing of the solid motors include ultrasonic, microwave, and radiographic analysis. Radiographic analysis (X-ray) is the most effective and most common test technique for solid motors. However, the reliability of poststerilization X-ray inspection is degraded by the general inaccessability of the system after sterilization.

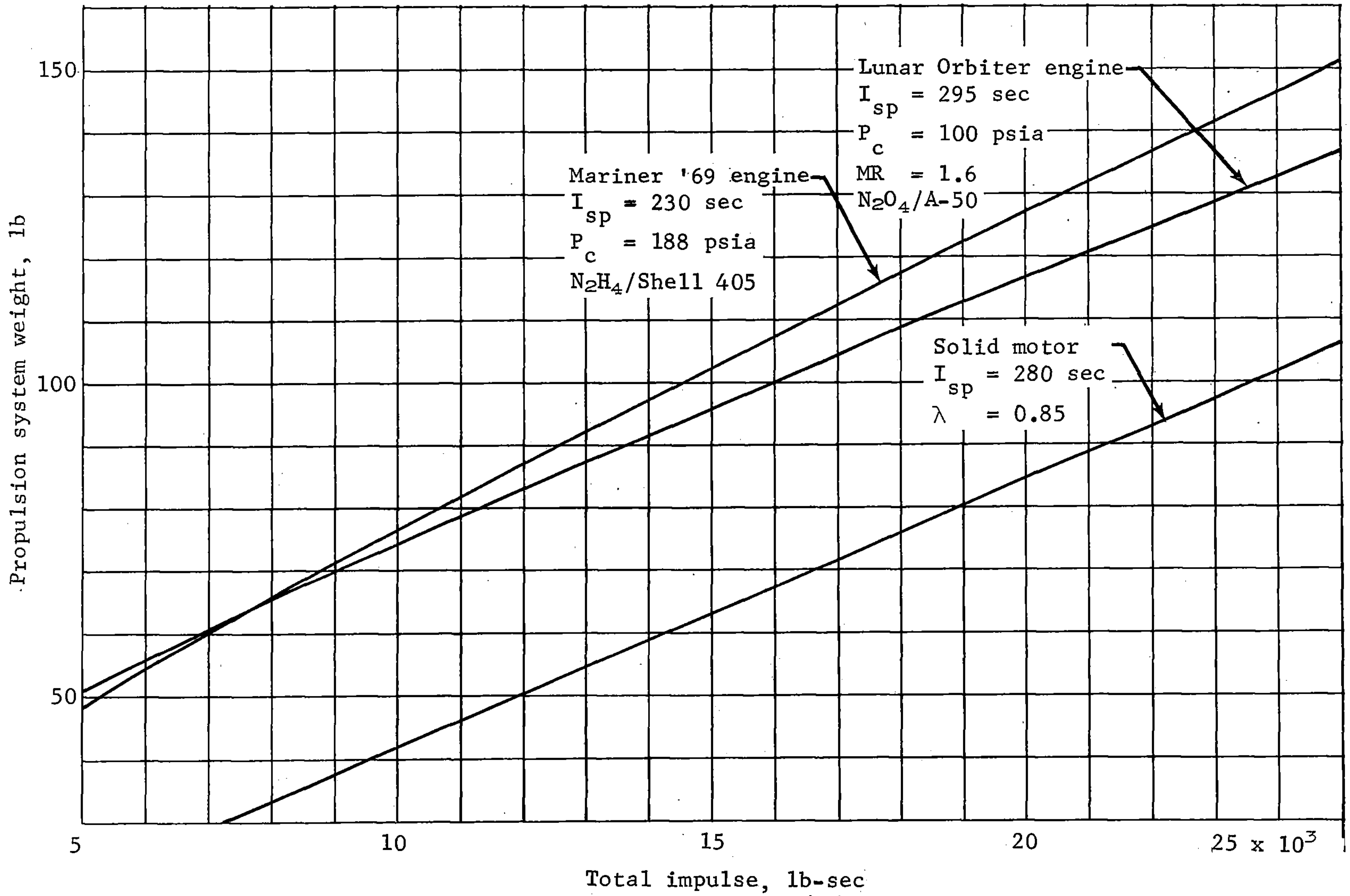


Figure 121.- Propulsion System Weight vs Total Impulse

It appears that solid rocket motor reliability would be degraded by sterilization. Such things as end unbonding, grain cracking, propellant/linear separation, insulation/wall separation, and nozzle seal leakage may be caused or aggravated by sterilization and are difficult to adequately check.

Deflection propulsion subsystem.- The deflection propulsion system is required for both the 1972 flyby and the Venus/Mercury flyby missions. The difference between the two missions is the magnitude of the velocity vector to be supplied to the capsule. These velocity increments are 50 and 93 m/sec, respectively.

The propulsion subsystem, figure 122, is a monopropellant hydrazine system using the Mariner '69 rocket engine to provide thrust. This engine was selected for the deflection system because of its state of development, and ability to provide the required velocity increment of 50 m/sec to the capsule without changes. Pressurization is accomplished by blowdown of gaseous nitrogen at a blowdown ratio of 3:1. The overall system characteristics are summarized in table 89.

Squib-operated valves (with redundant squibs) isolate the propellant and pressurant before system activation. On activation, the ordnance valves open the pressurant and propellant lines and start the blowdown cycle. A squib-operated valve reseals the propellant system at the end of the deflection maneuver to ensure a positive, controlled shutdown of the engine.

Table 90 presents the weight statement for the deflection propulsion subsystems.

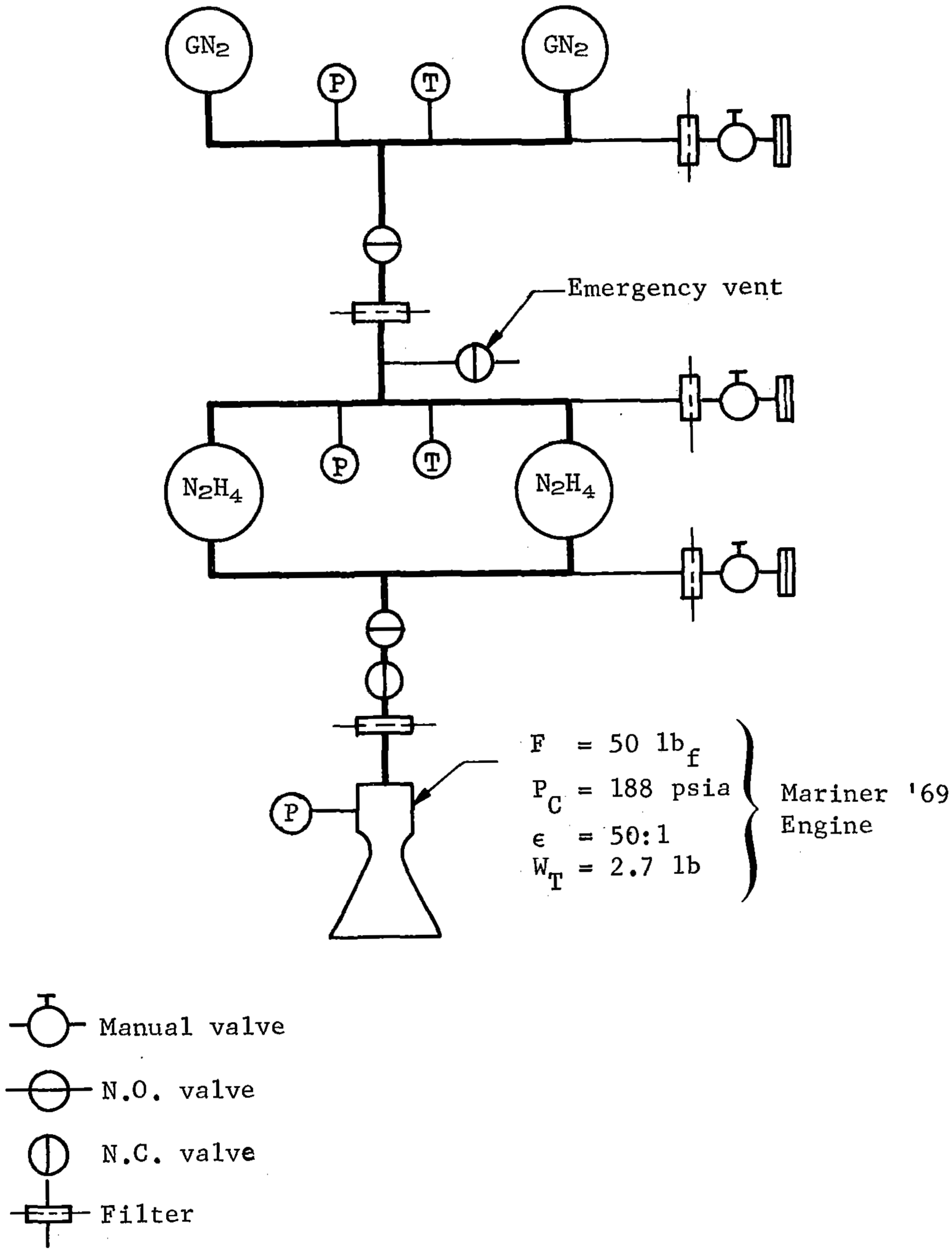


Figure 122.- Capsule Deflection Propulsion Subsystem

TABLE 89.- CAPSULE DEFLECTION PROPULSION SUBSYSTEM CHARACTERISTICS

Parameter	Mission	
	1972 flyby	Venus/Mercury flyby
Engine thrust (initial), lb_f	50 ± 2.5	50 ± 2.5
Engine thrust (final), lb_f	22 ± 1.1	22 ± 1.1
Maximum chamber pressure, psia	188	188
Average delivered specific impulse, sec	227 (230 max)	227 (230 max)
Nozzle expansion ratio	50:1	50:1
Engine weight, lb	2.7	2.7
Usable fuel weight, lb	22.4	46.6
Fuel contingency, lb	0.6	2.3
Fuel tank weight (2), lb	2.7	3.5
Volume per fuel tank, cu in.	355	760
Fuel tank outside diameter, in.	8.8	11.4
Pressurant weight, GN_2, lb	0.4	0.8
Pressurant tank weight (2), lb	1.3	2.1
Pressurant tank volume, cu in.	189.5	450
Pressurant tank outside diameter, in.	7.1	9.6
Operating temperature range, min./max., °F	40/80	40/80
Storage temperature range, min./max., °F	40/100	40/100
Initial fuel tank operating pressure, psia	300	300
Initial pressurant tank storage pressure, psia	307	307

TABLE 90.- CAPSULE DEFLECTION PROPULSION SUBSYSTEM WEIGHT STATEMENT

Item	Weight, lb	
	1972 flyby	Venus/Mercury flyby
Fuel, total	23.0	48.9
Fuel tanks	2.7	3.5
Pressurant (GN_2)	0.4	0.8
Pressurant tanks	1.3	2.1
Rocket engine	2.7	2.7
Ordnance valves	3.2	3.2
Filters	.8	.8
Instrumentation	1.2	1.2
Fill and drain assemblies	3.3	3.0
Tank and engine supports	8.6	10.5
Tubing	2.0	2.0
Total system weight	49.2	78.7

Thrust vector control is accomplished by spin stabilization of the capsule. Two fuel tanks and two pressurant tanks are located symmetrically about the capsule spin axis for balance. After completion of the deflection maneuver, the deflection propulsion system is separated from the entry capsule and enters the atmosphere on a different trajectory from that of the entry capsule.

Solid and bipropellant subsystems were considered for the deflection propulsion system. However, the small total impulse requirement and the mission weight margin allowed selection of a monopropellant system.

The blowdown monopropellant hydrazine deflection system has simplicity, an advanced stage of development, and flexibility. Blowdown pressurization eliminates the regulator and its associated failure modes from the system. The Mariner '69 engine will have flown to Mars and will be tested in flight before the Venus launch. Therefore, the selected system will have a qualified and fired engine and will eliminate development risk.

THERMAL CONTROL

The thermal control requirements, except those associated with the inflated balloon, are presented in this section for the orbiter, flyby, and Venus/Mercury missions. The requirements are defined by the three basic phases of each mission, i.e., cruise, coast, and floating in the Venusian atmosphere.

Temperature control during the cruise phase can be accomplished most simply by coatings and electrical heaters. A maximum power of 38 W is supplied by the spacecraft solar panels. The coast phase temperatures can be controlled passively with thermal coatings. The expected mild temperatures of the Venus atmosphere at the design float altitude will result in reasonable component temperatures in the gondola. A solar-reflecting coating is used on the exterior of the gondola.

The following requirements and constraints were considered for this study:

1) Temperature limits - Operating and nonoperating temperatures are given in the following tabulation. The most significant operating limit is the battery.

Component	Operating Limit, °F	Nonoperating limit, °F
Electronics	15 to 140	-35 to 160
Battery	50 to 90	40 to 90
Balloon gas	[a] 50 to 90	-65 to 120

[a] At initiation of inflation.

2) Power duty cycle - The power duty cycles and component power requirements are covered in the power subsystem.

3) Environment -

   a) Sterilization - During sterilization all surfaces of the capsule will be subject to a dry heat of 257°F (275°F for qualification);

   b) Park orbit, acquisition, and midcourse correction - The amounts of Earth albedo and emission received vary along with the sun orientation;

c) Cruise - The solar flux will vary from 444 Btu/hr-ft^2 at Earth to 850 Btu/hr-ft^2 at Venus. The orbiter and flyby missions will have a solar flux of 900 Btu/hr-ft^2 at the closest point of the trajectory to the sun;

d) Orbit - The capsule will be subject to planet albedo, emission, and shielding from the sun;

e) Coast - The entry capsule will be subject to the solar space environment for 10 to 68 hr before entry;

f) Entry - Entry will impose acceleration forces up to 300 g. The backface temperatures of the forward and aft ablators will reach 400°F;

g) Parachute descent - The local atmosphere can reach -66°F during the 15-minute descent. No appreciable aeroheating is experienced;

h) Floating in the Venus atmosphere - The model atmosphere temperatures range from 59 to 84°F at the design float altitude. The atmosphere models are discussed in detail in appendix A. Temperature variation due to weather is ±9°F, which results in a design atmosphere temperature range of 50 to 93°F. Floating in and out of the clouds will vary the solar and infrared radiation on the BVS.

Orbiter Mission

Cruise analysis.- The solar flux varies from 444 Btu/hr-ft^2 at Earth to 900 Btu/hr-ft^2 at the closest point of the trajectory to the sun, to 850 Btu/hr-ft^2 at Venus. Figure 123 shows the capsule temperature change going from Earth to Venus using coatings only. Other combinations of coatings could change the temperature level, but would not change the range significantly. This temperature range exceeds some of the component temperature limits, thus it is necessary to employ some method of control.

Figure 124 shows four methods studied for a similar Venus mission. The results are applicable to this mission. The first system uses electrical heaters on the gondola and the propulsion subsystem. A radiation shield is added to the propulsion subsystem to reduce the amount of electrical power required.

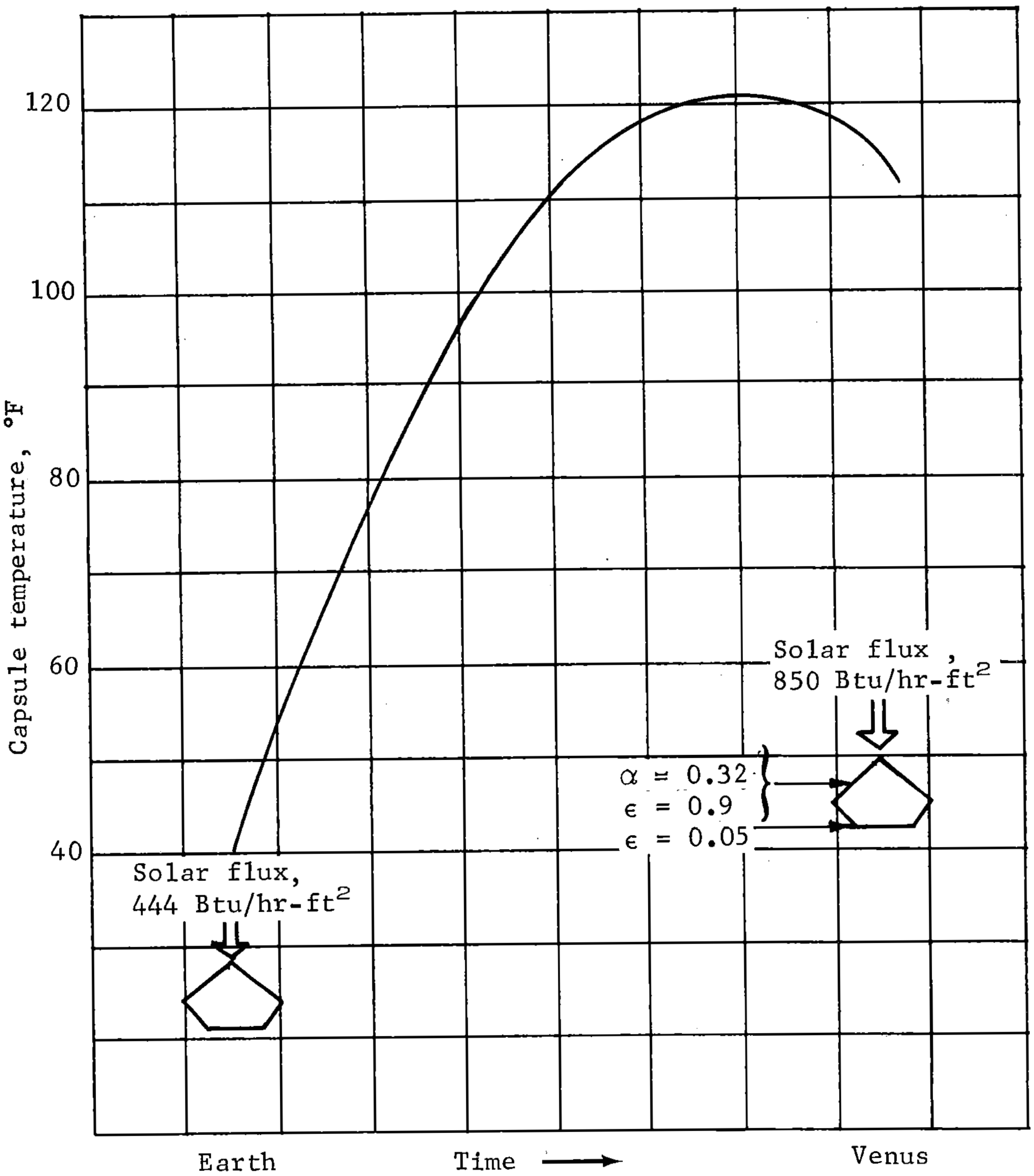


Figure 123.- Passive Cruise Control

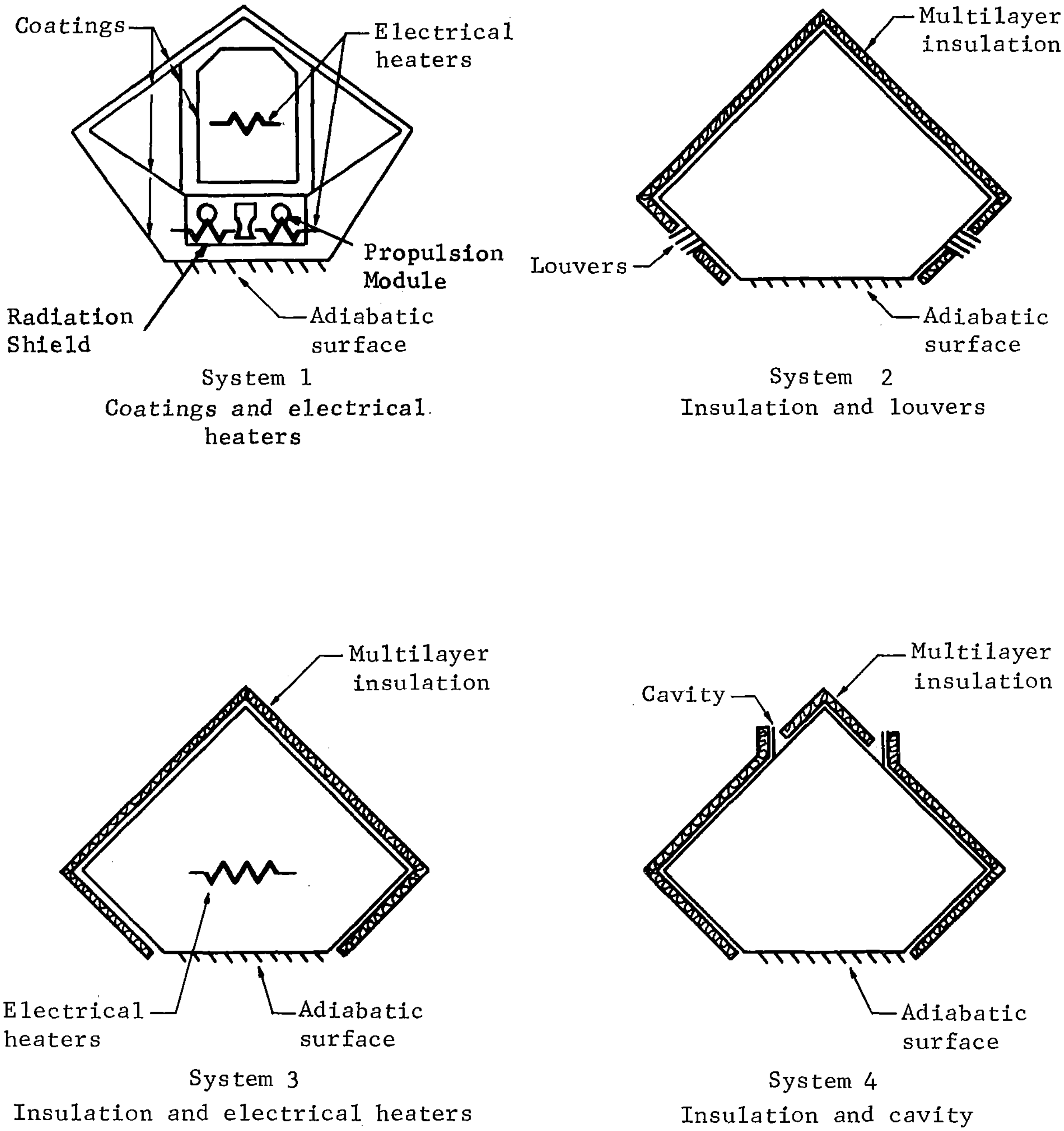


Figure 124.- Possible Thermal Control Systems for Cruise Mode

Each surface has a specific thermal coating to control the capsule temperature. The three other systems use multilayer insulation on the exterior surface of the biocanister, resulting in a small capsule temperature gradient. In system 2, control is obtained from louvers that reject heat at Venus. In system 3, heat is added to the capsule by electrical energy supplied by solar panels. In system 4, heat is added to the capsule by having an adjustable flap to absorb solar energy. System 3 and 4 are sized so that no heat is added at Venus.

One important parameter affecting all systems is the range of coating properties that can be obtained. Data from Mariner '67 indicate a 200% degradation of white paints. This degradation increases the solar absorptivity which, in time, increases the thermal control requirements. Some types of conversion coatings are stable. Flight data of Pegasus indicate that its conversion coating, MTL-3, is performing satisfactorily. Two solar reflective conversion coatings are Fairchild DEV-26-70, $\alpha/\epsilon = 0.5$, and DEV-26-20, $\alpha/\epsilon = 0.67$ (ref. 15). Conversion coatings chemically similar to MTL-3 can have values of α/ϵ of about 0.6. The lowest value of α/ϵ considered in our study is 0.6. A maximum value of α/ϵ equal to 2.8 was based on an outer covering temperature of 800°F at Venus.

Another parameter affecting the cruise mode is the performance of multilayer insulation. Considering the wide spread in performance data of multilayer insulation, an overall performance variation of ±50% was used. A tolerance of ±25% was used for performance variation from point to point in a given installation. A nominal value of 0.06 was used for the effective emissivity of each surface. This value accounts for attachments, joints, and perforations and was determined from a full-scale test of a 19-ft-diam Voyager thermal model (ref. 16). Martin Marietta test results are very similar to the results of General Electric Company (ref. 17) as shown in the following tabulation

Item	Martin Marietta	General Electric
Covering	2-mil Kapton	2-mil Kapton
Shields	10 layers 1/4-mil Mylar, aluminized both sides	22 layers 1/4-mil crinkled Mylar, goldized one side
Separators	2 layers silk mesh	None
Performance, Btu/hr-ft^2	0.40 at 55°F	0.44 at 55°F
Weight, lb/ft^2	0.078	0.075

The study was based on the General Electric insulation system. Mylar physical properties become marginal at 300°F. Thus shields which will be subjected to temperatures of 300°F or higher may be required to be made of Kapton. No weight penalty is assumed for using Kapton instead of Mylar in this study.

System 1 - electrical heating of the probe and propulsion system: Power for the heater is taken from the solar panels on the spacecraft. For each pound of solar panels, 6 W are produced (ref. 19); this energy density includes structural support for the solar panels. Wire mesh blanket-type heaters controlled by series-parallel bimetallic thermostats are used.

The system weighs 6 lb of which 2.5 lb are for heaters and controls. The other 3.5 lb are for the solar panels that are required to produce 21 W of thermal control power.

System 2 - insulation and louvers: Multilayer insulation is used on the exterior surface of the biocanister so that the louvers will be opened at Venus and closed at Earth. The effective emissivity of closed louvers is 0.1 and of opened louvers is 0.65 (ref. 18). A louver weight of 1 lb/sq ft was taken from reference 19. Using the worst combination of expected variation of insulation performance, it was determined that the louvers could not limit the capsule to a 10°F temperature rise. Allowing the temperature to range from 40°F at Earth to 90°F at Venus, an optimum system weight of 34.5 lb was determined. The weight includes 13.5 lb of insulation, which has eight shields on the sun-oriented forward surface and 40 shields on the aft surface. A louver area and weight of 21 sq ft and 21 lb, respectively, is needed. A solar absorbing surface, $\alpha/\epsilon = 2.7$, is required on the outer covering facing the sun.

System 3 - insulation and electrical heaters: Multilayer insulation is used on the exterior surface of the biocanister so that added heat is required at Earth but not at Venus. Electrical power is obtained from solar panels on the spacecraft at 6 W/lb.

Using the worst combination of expected variations in insulation performance, the system was optimized assuming the capsule is isothermal. Holding a constant temperature of 60°F from Earth to Venus, an optimum system weight of 24 lb is required, consisting of 6.1 lb of solar panels (36.6 W maximum) and 2.8 lb of heaters and controls. The insulation weighs 15.1 lb with 30 shields on the forward position of the biocanister and four shields on the aft portion. The insulation outer covering requires an α/ϵ value of 0.63.

System 4 - insulation and solar cavity: Changing system 3 to obtain heat directly from the sun instead of from the solar panels will reduce the system weight. Solar energy can be absorbed and controlled by having an absorbing surface or cavity with an insulated movable cover. The cover, being similar to an insulated louver system, can vary the solar heat absorbed. The system optimization is the same as for system 3. For a constant temperature of 60°F the system weight is 15.5 lb with 15.1 lb of insulation and 0.4 lb for the 0.4 sq ft of cover area.

Selection: System 1 is the recommended system for the following reasons:

1) It is the simplest system, consisting only of heaters and coatings;

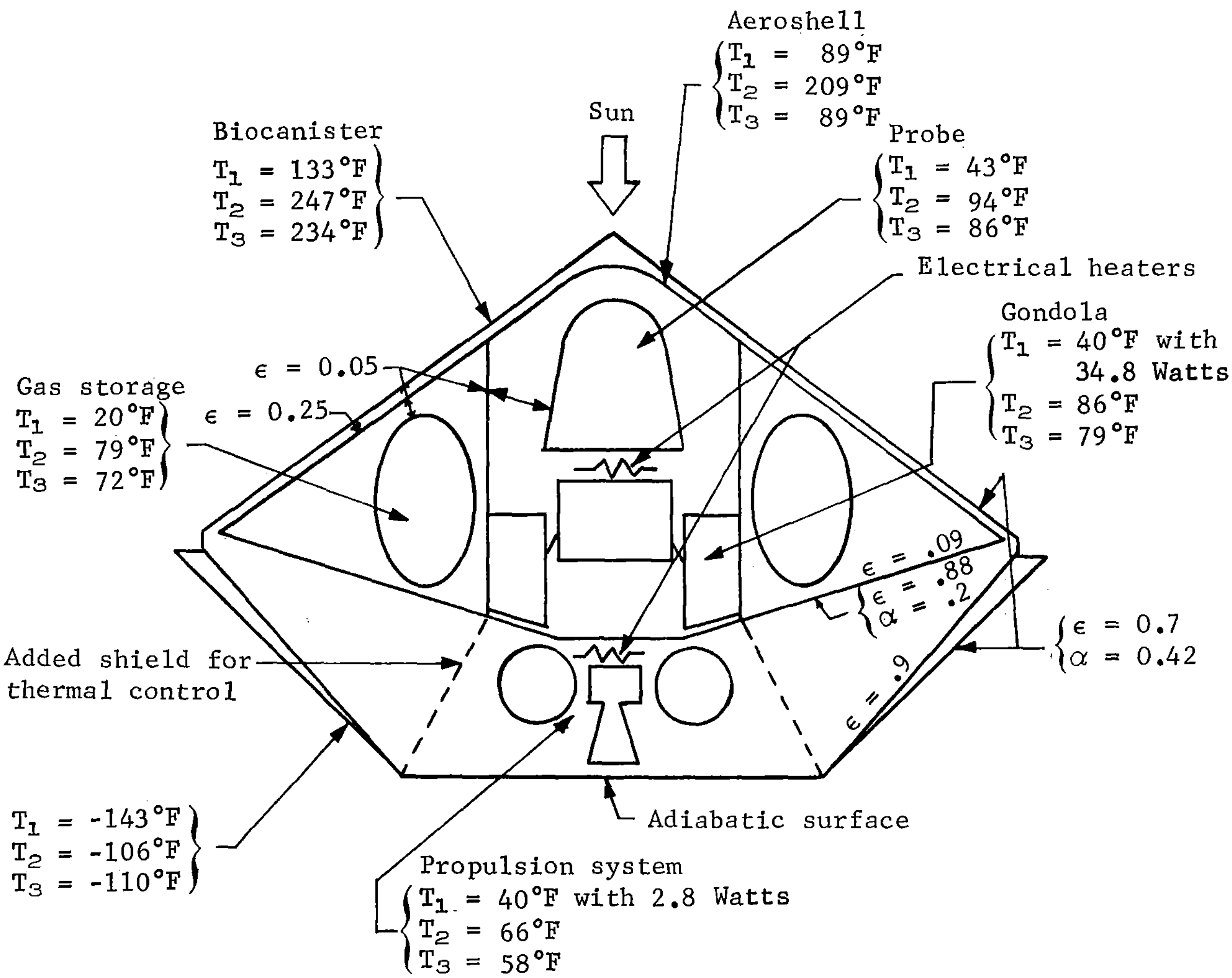
2) It is the lightest system;

3) It has the minimum number of development problems.

Figure 125 shows the thermal control system for Venus BVS cruise phase and the resulting temperatures during cruise. A maximum of 34.8 W is required in the gondola and 2.8 W in the propulsion system. The control for each wire mesh heater blanket is by series parallel bimetallic thermostats. The power is supplied by the solar panels on the spacecraft. A weight of 1 lb/6 W results in a weight of 6.3 lb for increasing the solar panels for the thermal control system.

The external coating on the biocanister is a conversion coating with α/ϵ equal to 0.6. Low emissivity surfaces are either polished metal or an aluminized or goldized film bonded onto the surface. The outside surface of the aft ablator is painted white -- a requirement for the coast phase. Only the shield around the propulsion subsystem is an added surface for thermal control. The resulting temperatures during cruise are shown in figure 125. The selected system will keep the propellants from freezing and the battery within its nonoperating range of 40 to 90°F.

During such maneuvers as midcourse correction, the loss of electrical energy is possible because of spacecraft orientation. The maximum temperature change of the propulsion or gondola during a one-hour reorientation is -5°F.



T_1 is temperature at Earth
T_2 is temperature at closest point to Sun (0.7 A.U.)
T_3 is temperature at Venus

Location	Solar flux, Btu/hr-ft^2
Earth	444
Closest point to Sun	900
Venus	850

Figure 125.- Cruise Phase Thermal Control

Orbit analysis.- The capsule has the same sun orientation during orbit as it does during cruise. Planet albedo and emission raise the temperature of the gondola and probe less than 1°F and the balloon gas storage tanks less than 2°F. The eccentric orbit and the 25-hr orbit period allow the capsule to return to the steady-state temperatures shown in figure 125 during each orbit. Venus shades the capsule from the sun for approximately 50 minutes.

Coast analysis.- During the coast phase, the biocanister and propulsion system are separated and the spinning entry capsule is oriented with the sun on the aft surface. The aft surface is coated with a white paint and the forward aeroshell is coated with an aluminum paint. All surfaces have the same coating as analyzed during the cruise mode. The temperature history for the 10-hr duration is shown in figure 126. Because coatings are adequate to provide thermal control, no other systems were considered.

Entry analysis.- The entry heating results in a rapid temperature **rise of the** backface of the ablator surfaces. Separation of the probe from the aeroshell occurs seconds after the backface reaches 400°F. The short exposure of the probe to this temperature will increase the temperature of the gondola less than 1°F. Thus, no additional control is required for entry heating.

Parachute descent analysis.- The maximum descent time is 14.8 minutes. The analysis considered forced convection on the exterior surfaces of the gondola and gas storage tanks. Free convection to the local atmosphere was assumed for the internal surfaces of the gondola because the gondola has small ventilation louvers. Figure 127 shows the temperature history of several components during descent. The resulting temperatures show no need for additional control during this phase of the mission.

Gondola analysis in the Venus atmosphere.- Estimated thermal properties of the Venus atmosphere are listed in table 91. The gondola was evaluated for both the hot and cold extremes. The gondola is vented so that free convection with local temperature atmosphere is possible. The outer surfaces of the gondola have been coated with a white paint to reduce the solar absorption. Possible degradation of the white paint has not been analyzed. The radiation shield on the interior surfaces of the gondola did not significantly reduce the component temperatures and will not be used. Figure 128 shows steady-state temperatures of the gondola with the programer powered with 4 W. Power peaks will increase the temperature of some components but steady-state temperatures will be reached during low power levels.

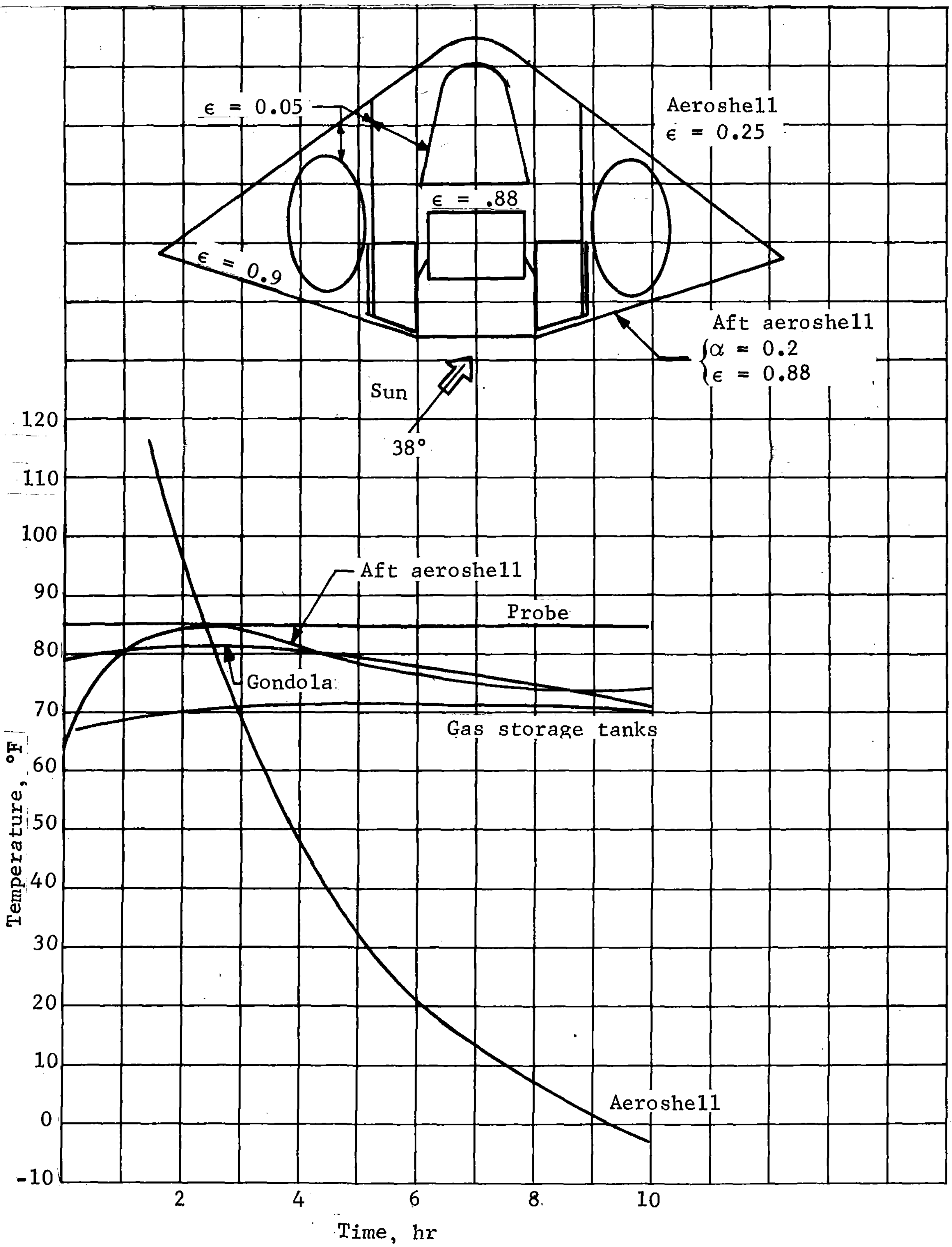


Figure 126.- Coast Phase Thermal Control

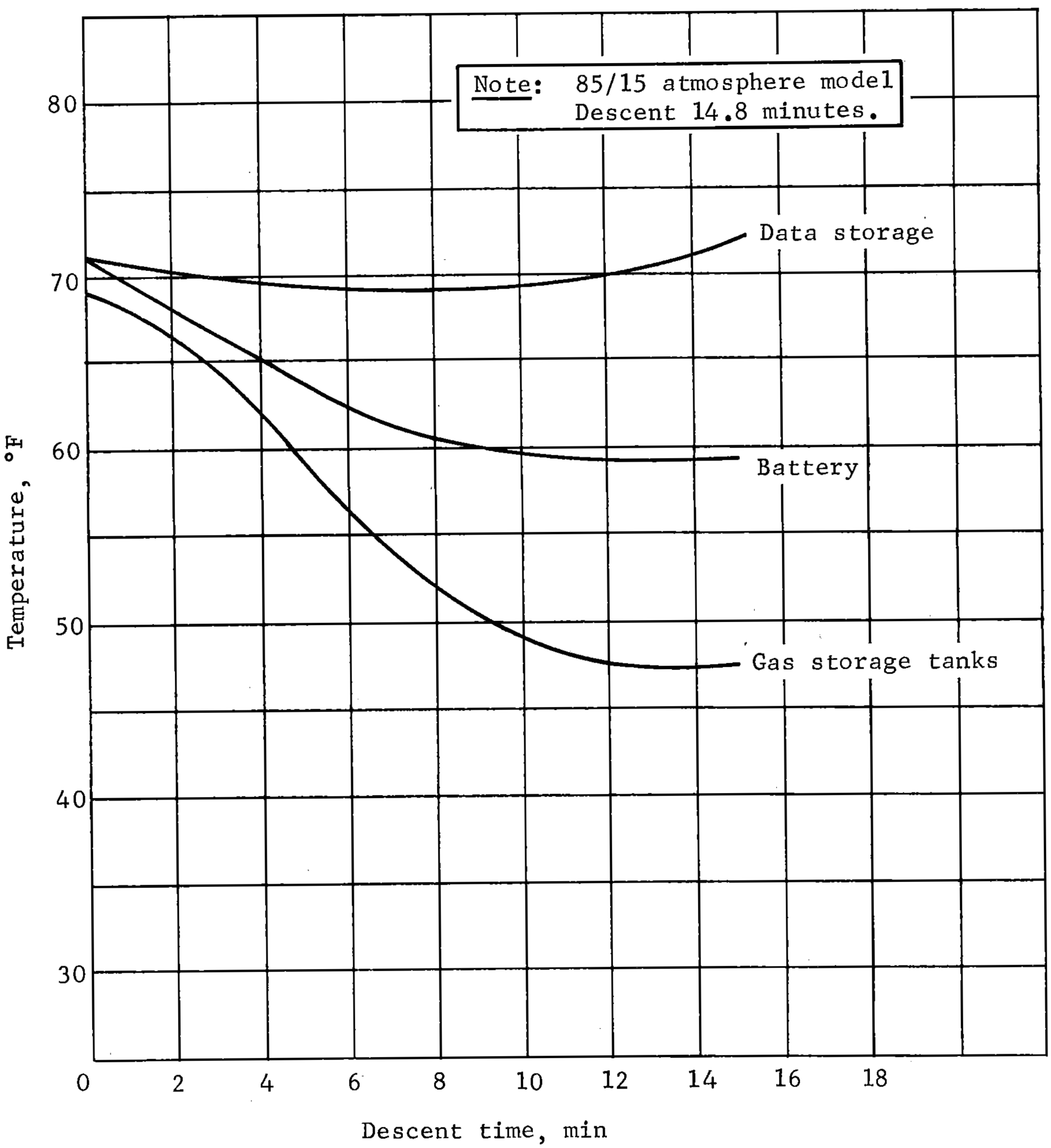


Figure 127.- Component Temperatures During Descent

TABLE 91.- THERMAL PROPERTIES OF VENUS ATMOSPHERE

Parameter	Range of values	Nominal Conditions: In clouds	Nominal Conditions: Out of clouds	Cold extreme	Hot extreme
Atmosphere ($\%CO_2/\%N_2$)	85/15 to 95/05	90/10		85/15	95/05
Float altitude, ft	$190\ 300\ {}^{+1310}_{-1640}$	190 300		188 660 $(t_a = 59°F)$	191 610 $(t_a = 84°F)$
Cloud top altitude, ft	210 000 ± 60 000	>190 300	<190 300	<188 660	<191 610
Cloud top temperature, °F	-22 ± 27	-22		-49	5
Atmospheric temperature (t_a), °F	Function of atmosphere and altitude, ±9	70		50	93
Sun angle, deg	0 to 90	0		90	0
Cloud solar transmissivity	0 to 1.00	0.14		----	----
Cloud emissivity	1.00	1.00		1.00	1.00
Atmospheric solar transmissivity	0.85 to 1.00	0.93		0.85	1.00
Atmospheric emissivity	0.20 to 1.00	0.25		0.20	1.00
Albedo	0.50 to 0.90	0.76		0.50	0.90

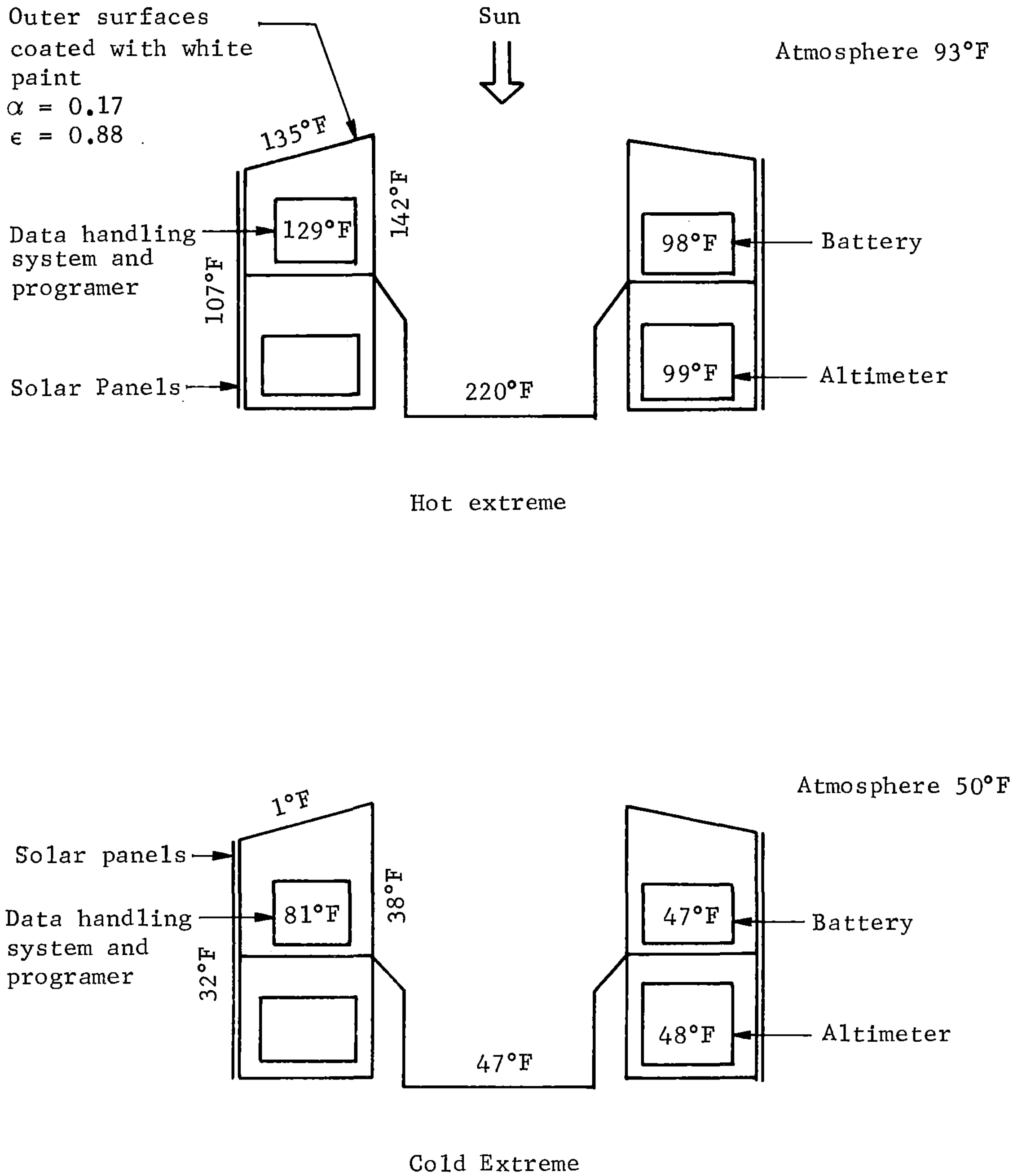


Figure 128.- Gondola Floating in the Venus Atmosphere

One pound of phase change material (tetracosane) including packaging is used to absorb the high energy peak of the S-band transmitter. Tetracosane is a paraffin with a melting temperature of 124°F and a heat of fusion of 109 Btu/lb. Figure 128 shows that the battery exceeds its upper temperature limit of 90°F. Cooling the battery below local atmospheric temperature is costly in weight. Thus, it may be desirable to take the loss of efficiency in operating the battery above 90°F. Fins can be added to battery to hold its temperature close to the atmospheric temperature, if battery efficiency requires it.

Flyby Mission

Most phases of the flyby mission are identical to those of the orbiter mission. The major difference is in the coast phase. The capsule temperatures (fig. 129) during the coast phase reflect the change in sun orientation and length of coast time. The coatings are the same as those used during the orbiter mission, thus, the thermal control design for both missions is the same.

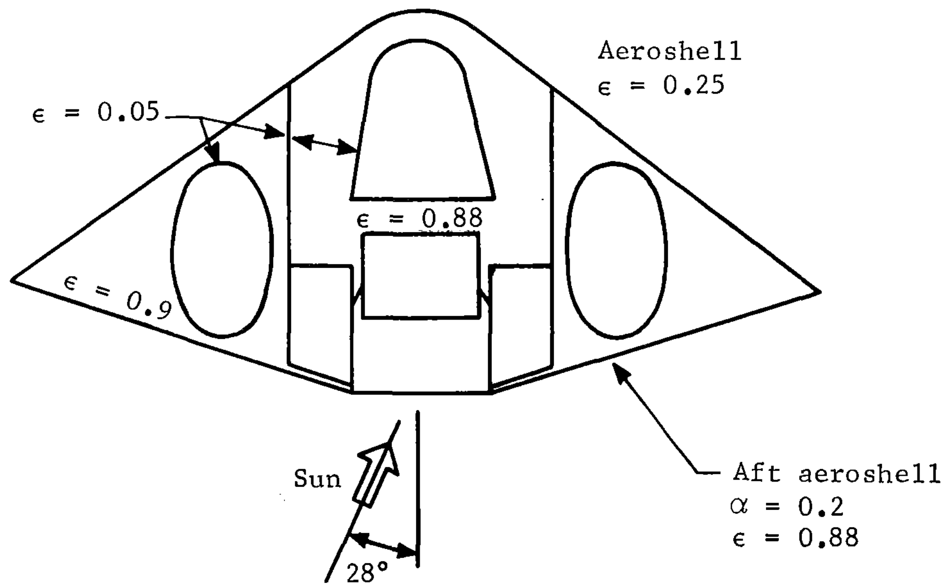
Venus/Mercury Mission

The Venus/Mercury flyby mission has several factors that alter the results from the orbiter and flyby missions. Floating on the dark side of Venus reduces the balloon and gondola temperatures. The new sun orientation during coast requires a change in the entry capsule coatings, which affects the cruise mode temperatures.

During coast phase the sun orientation will require white paint on the forward aeroshell and a low emissivity coating on the aft aeroshell. The resulting capsule temperatures during cruise are shown in figure 130. A 37 W power supply to the gondola and propulsion system will maintain their temperature at 40°F when the spacecraft is near Earth. During cruise, the capsule does not approach 0.7 A.U. from the sun before Venus encounter as in the orbiter and flyby missions.

Transient analysis during acquisition phase and midcourse correction showed the temperature change of the probe and gondola was less than 5°F for a 1-hr duration. The temperature history for the 68-hr coast period is shown in figure 131.

The hot extreme will be less severe on the dark side of the planet because of the absence of any solar radiation. The cold extreme will be the same as shown in the orbiter mission.



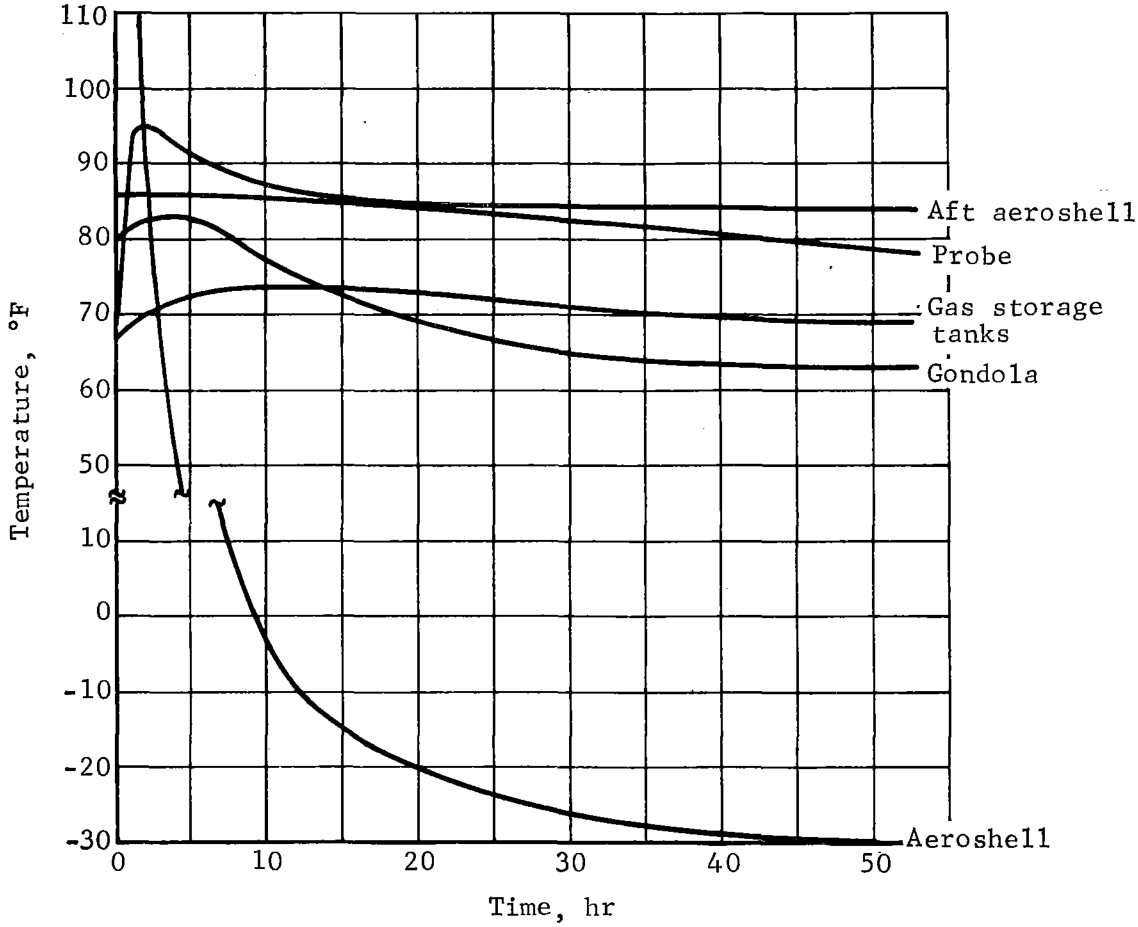
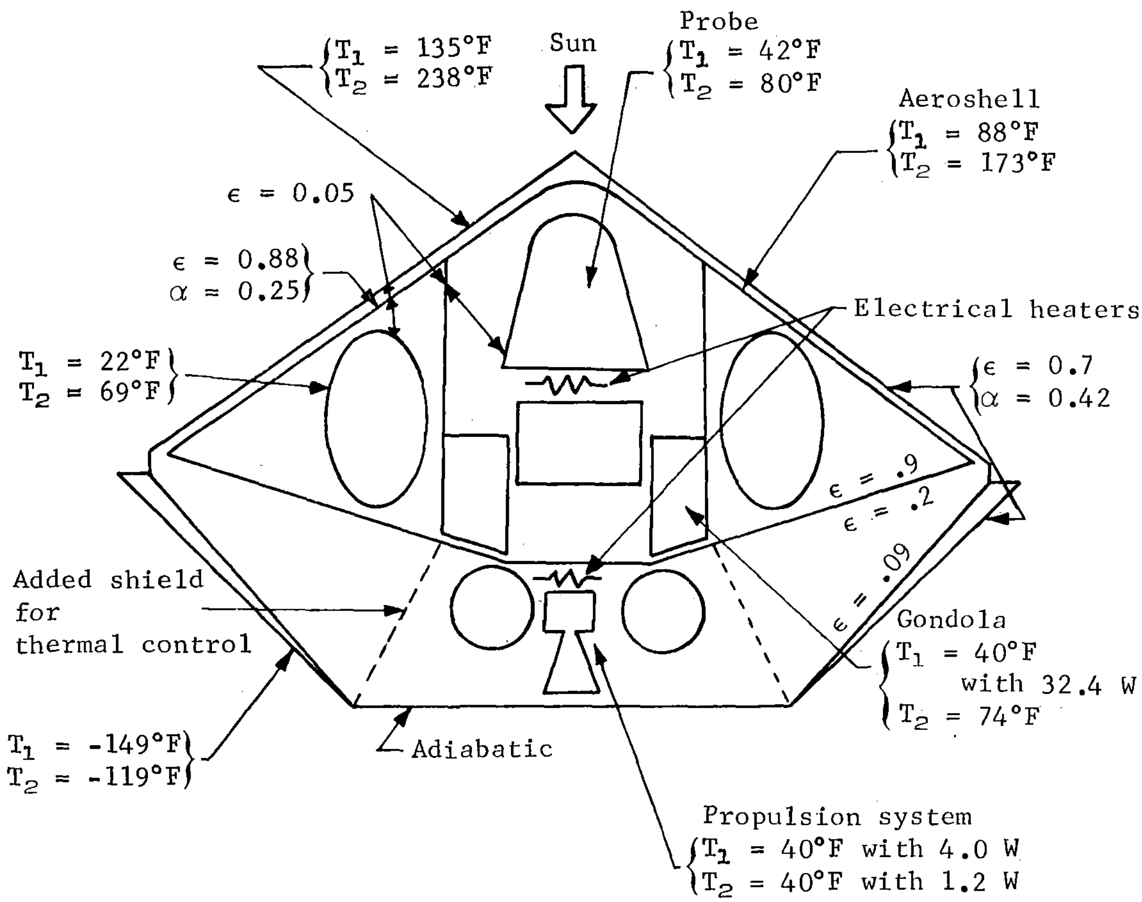


Figure 129.- Separation Phase Thermal Control



T_1 is temperature at Earth
T_2 is temperature at Venus

Location	Solar flux, Btu/hr-ft^2
Earth	444
Venus	850

Figure 130.- Cruise Phase Thermal Control

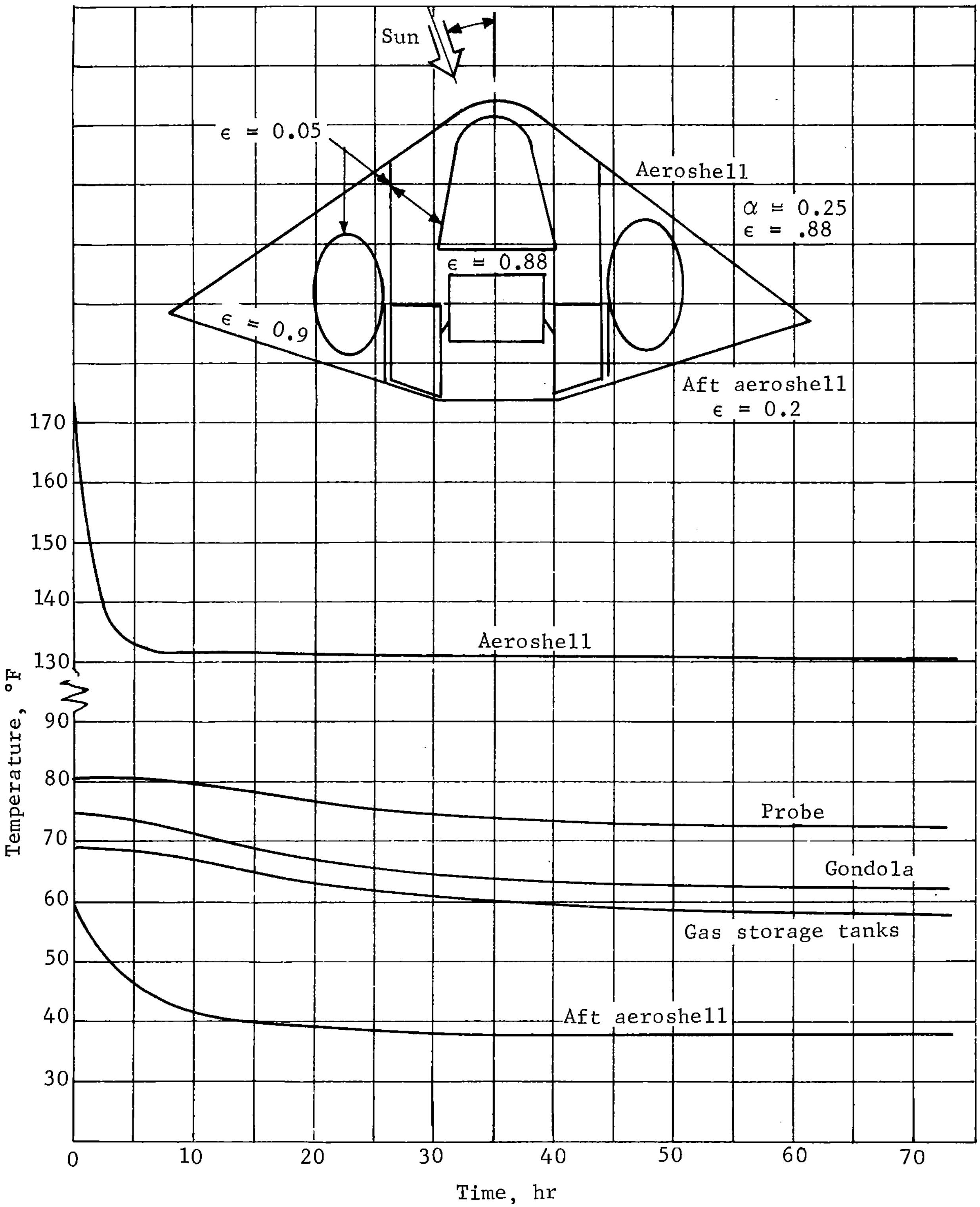


Figure 131.- Coast Phase Thermal Control

The hot extreme will reduce the data handling system from 129°F (as shown in fig. 128) to 125°F. The battery and altimeter temperatures will approach the atmospheric temperature of 93°F. The desired battery temperature will be exceeded, but it is desirable to accept the loss in battery efficiency rather than actively cool it below local ambient temperature. One pound of phase change material, tetracosane, will still be required on the S-band transmitter.

STRUCTURE AND MECHANISMS

For the three missions under consideration the structural concepts are identical except the aeroshell diameter is 7.0 ft for the Venus/Mercury mission as compared to 8.5 feet for the other missions.

The 8.5-ft-diam entry vehicle mounted in the biocanister, attached at eight places to the capsule adapter ring beam, is shown in figure 132. Each of the eight attach points contains a spring to supply separation velocity when the entry vehicle separates from the spacecraft. The aft section of the biocanister is attached to the spacecraft by V-band clamps. The large diameter ring of the aft biocanister mounts the total system to the Titan III transtage payload ring frame. The forward section of the biocanister is separated pyrotechnically to allow deployment of the entry vehicle. The 7.0-ft diameter entry vehicle for the Venus/Mercury mission is mounted in identical fashion with an accordingly smaller ring beam.

The entry vehicle showing all subsystems is shown in figure 133. The aeroshell is 8.5 ft in diameter, with a 55° cone half angle and 1.0-ft nose radius. The structure is aluminum ring frame, skin construction covered with ablator material. The afterbody mates with the forebody and is sealed with an O-ring seal to prevent the entry of hot gas into the capsule during the entry heat pulse. The aeroshell has a major ring frame with a short cylinder section that mates with the gas inflation system. Semimonocoque construction is carried through all the center structure. The gas inflation system section carries the four hydrogen tanks supported by trusses around the outer shell. All valving, filters, fittings, and tubing are located in this section and are released after balloon inflation.

The subsonic probe is structurally attached to the gas inflation subsystem and released by two pyrotechnic devices. The gondola is mounted on top of the gas inflation system.

The propulsion module for the orbital and flyby missions are shown in drawing sections B-B, and C, respectively. The two modules allow for a common attachment scheme to the afterbody.

The aeroshell for the Mercury/Venus mission is 7.0 ft in diameter but is structurally the same as the 8.5-ft-diameter capsule in general arrangement and design features. An inboard profile is shown in figure 134. There are no modifications to the BVS system within the aeroshell and base cover, and the structural interface remains unchanged. The conical afterbody is sharper for the 7.0-ft-diam capsule and the propulsion system is appreciably different with an attitude control, cold gas system incorporated into the propulsion module.

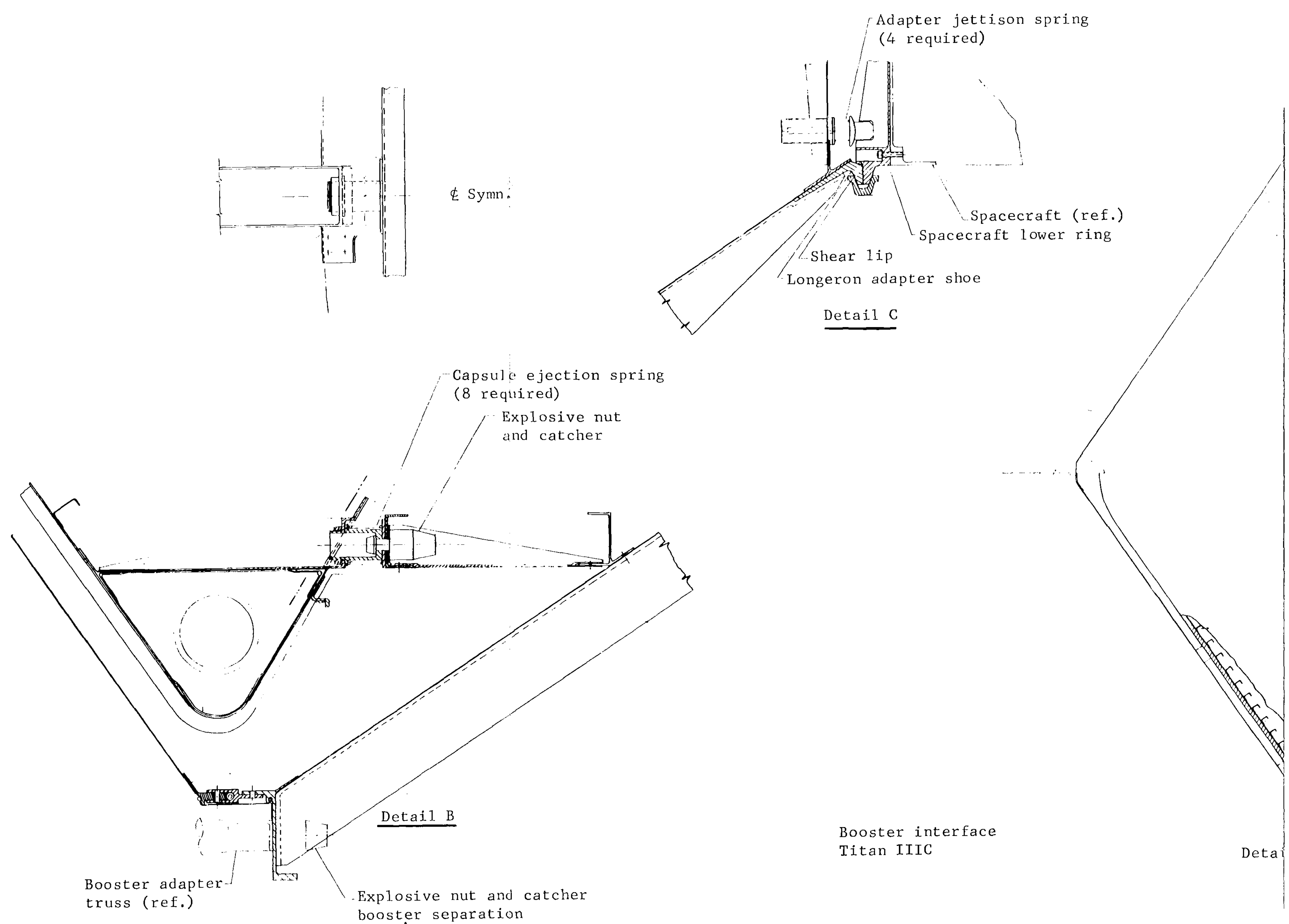
The afterbody has an ablator material covering the aluminum structure. An rf transparent window is required over the S-band antenna. The afterbody mates with the forebody and seals with an O-ring around its periphery at the base of the aeroshell major ring frame and separates pyrotechnically at the upper end of the BVS gondola.

The deflection propulsion system is supported on the afterbody and is separated pyrotechnically with jettison impulse provided by springs.

This capsule system is different from those of the orbiter and Venus flyby missions in that it has an active attitude control system (ACS). It separates with the deflection propulsion module. The ACS nozzles are supported off the propulsion module support structure by tubular beams that extend out toward the aeroshell periphery. The beams contact the afterbody at the periphery within receptacles, ball and socket type support, and are preloaded on installation. This assures rigid support during ACS operation and allows it to be totally jettisoned with the deflection propulsion system.

Heatshield

The ablative heatshield material selected for the orbital and flyby missions is ESA-5500(M). This is an elastomeric silicone-based material that uses carbon filler and reinforcement systems to maximize the surface temperatures at which it can operate without experiencing excessive recession rates. ESA-5500(M) is a a modification of the ESA-5500 material used to protect the high heating rate regions of the vehicle during the PRIME flight test program. The nature of the modifications result in improved char strength and, hence, shear resistance. Although ESA-5500(M) is not necessarily the optimum heatshield material for a Venus entry probe, its properties and response to the spectrum of environments to be encountered are relatively well-known because of its similarity to ESA-5500, which has been extensively tested both on the ground and in flight.



Adapter jettison spring
(4 required)
¢ Symn.
Spacecraft (ref.)
Spacecraft lower ring
Shear lip
Longeron adapter shoe
Detail C
Capsule ejection spring
(8 required)
Explosive nut
and catcher
Detail B
Booster adapter
truss (ref.)
Explosive nut and catcher
booster separation
Booster interface
Titan IIIC
Deta

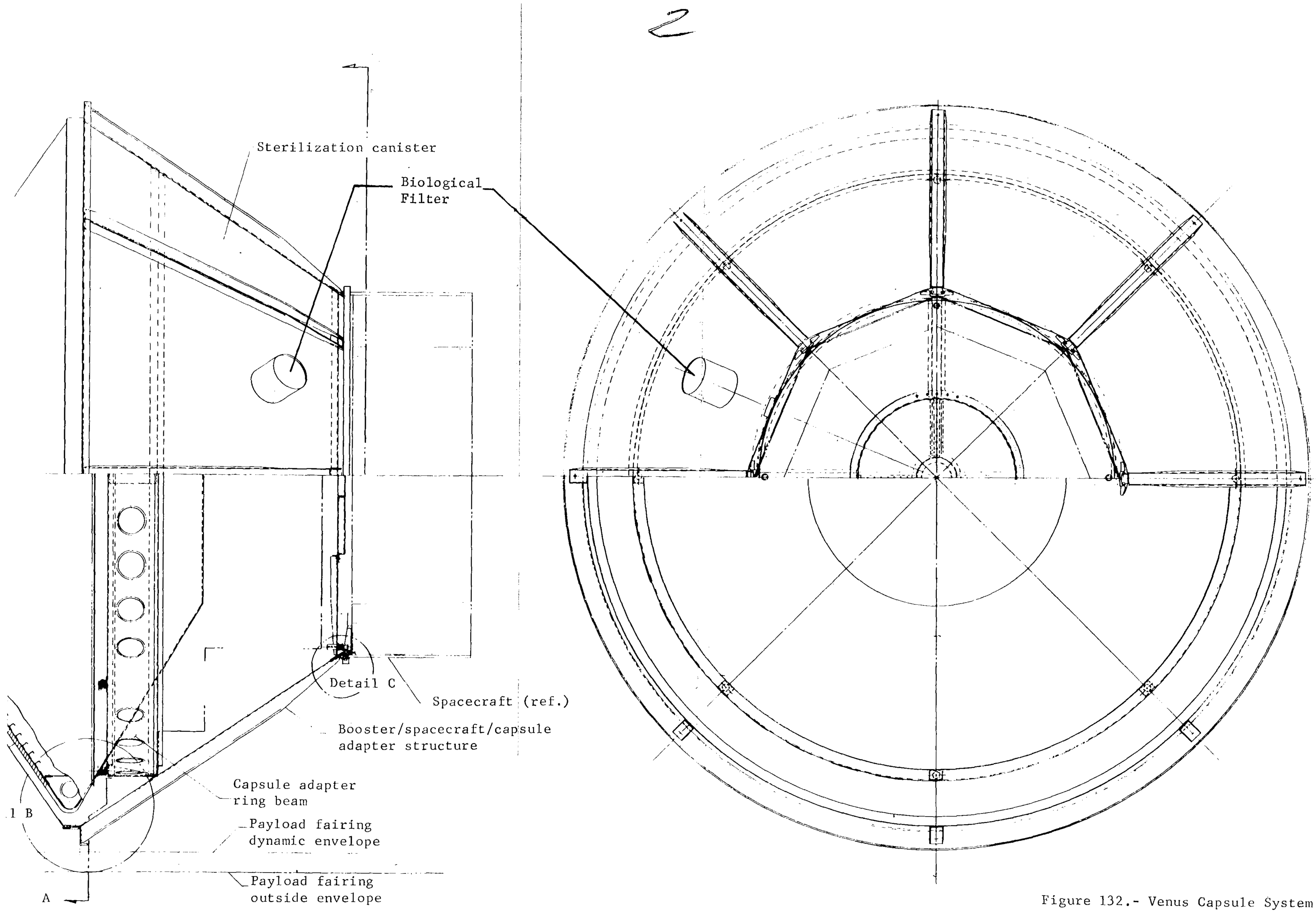


Figure 132.- Venus Capsule System

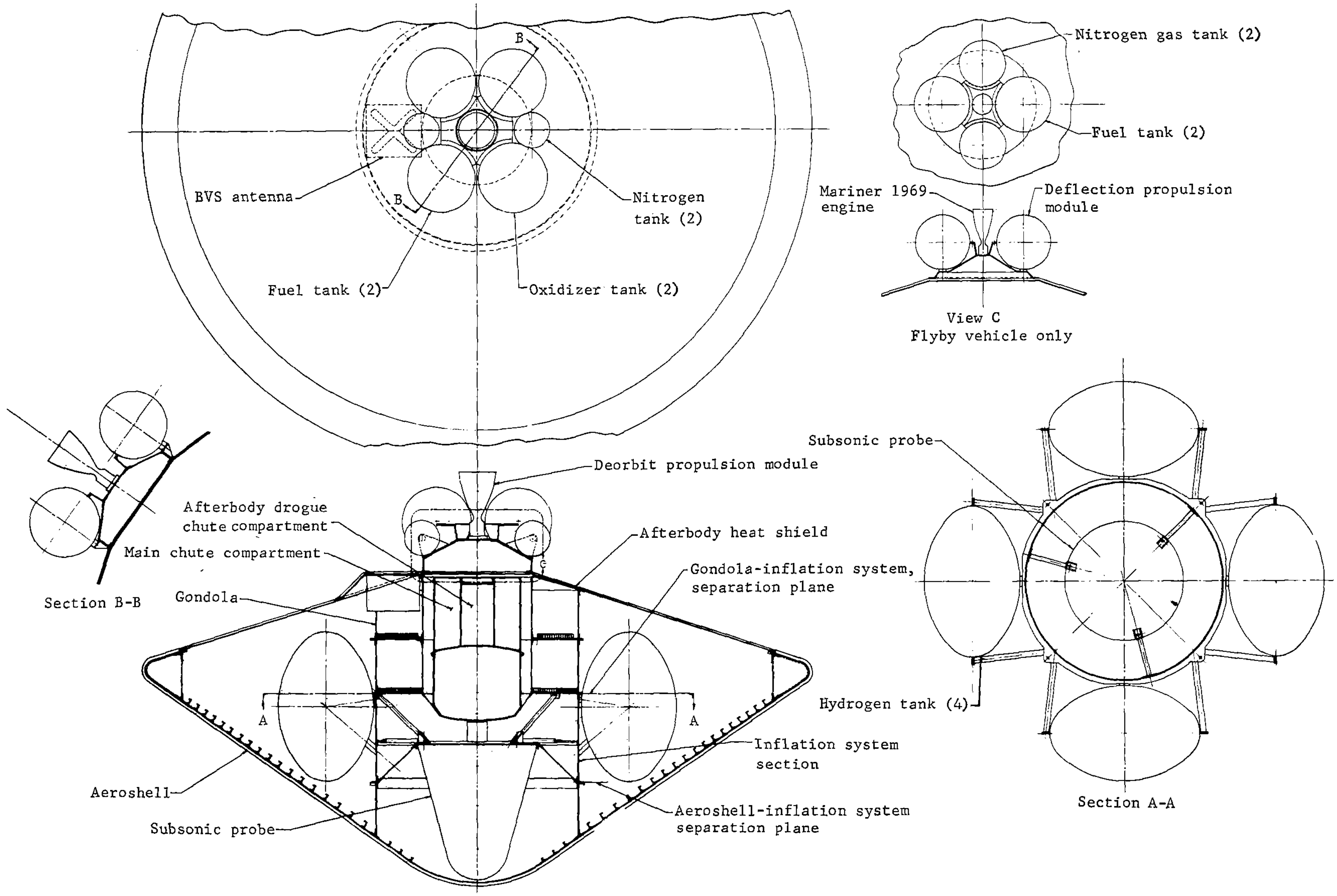


Figure 133.- BVS Entry Capsule

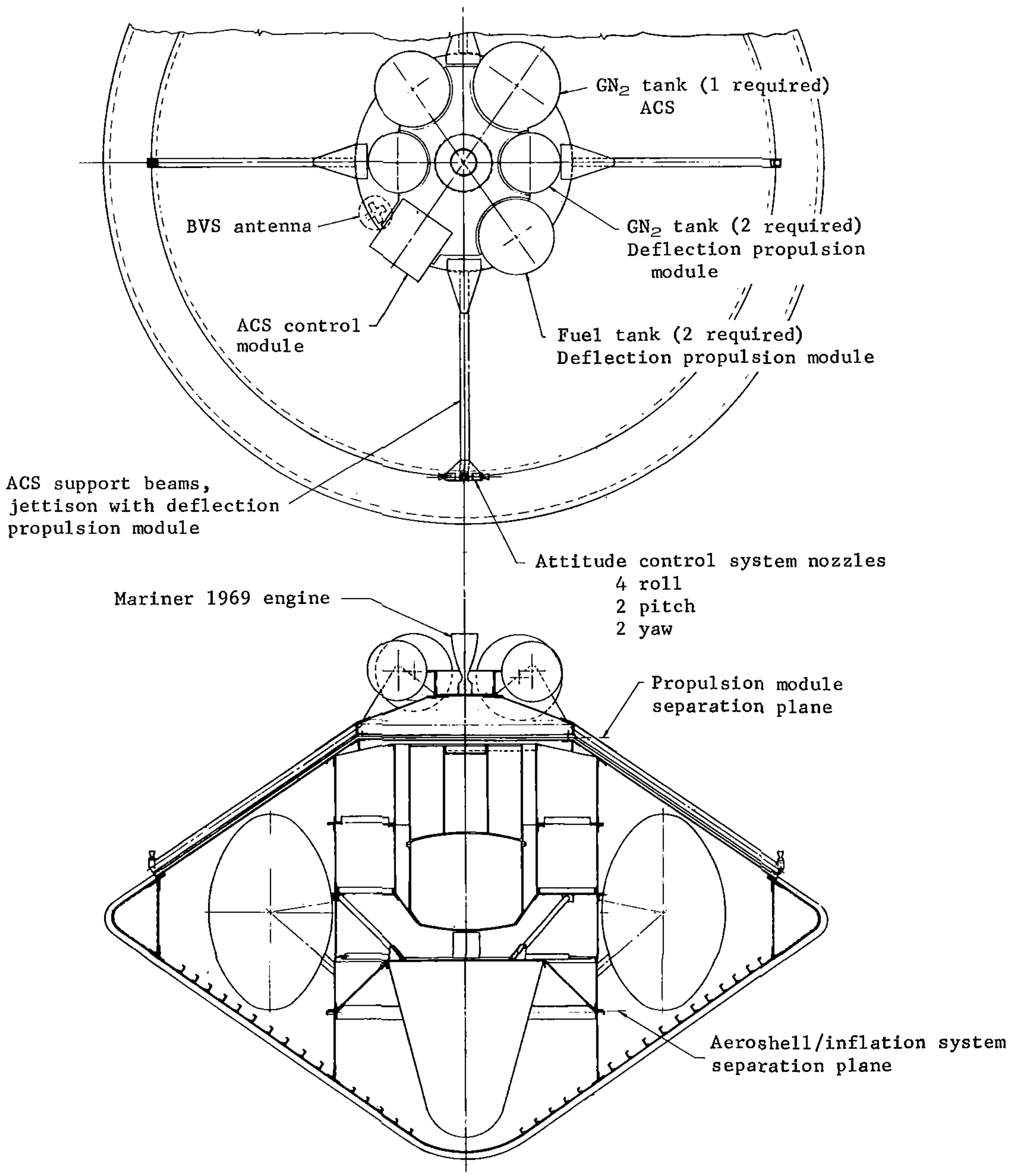


Figure 134.- BVS Entry Capsule, Venus/Mercury Swingby

The single environment that may be a problem for this or any material is the vacuum exposure associated with interplanetary transit. The magnitude of offgassing, the species release, and the effects of exposure on thermal and mechanical properties must be determined.

The thermal properties of ESA-5500(M) used in these heat shield design analyses are given in table 92. The safety factors and criteria applied to the analysis are given in table 93.

Since the entry velocity for the Venus/Mercury mission is approximately 44 000 fps, compared to 32 000 fps for entry from orbit, and 38 000 fps for entry from approach, the heat shield design difficulties arising from the uncertainty, level, and spectral distribution of the radiative heat flux are magnified in relation to the orbital and flyby missions. In addition, the viscous shear stress levels on the cone are increased to the point where it becomes doubtful if the ESA-5500(M) materials would be applicable for this mission and carbon phenolic becomes a more practical material.

The effects of the increased severity of the environment of the heat shield design and material selection can be summarized as follows:

1) Carbon phenolic becomes the prime material candidate because of its higher shear resistance and char strength;

2) Since carbon phenolic is not as efficient thermally as ESA-5500(M), the heat shield weight would increase from approximately 200 lb for the direct entry mission to 420 lb for the Venus/Mercury mission, if 8.5-ft-diam aeroshell and a 300°F design backface temperature criteria are retained. The reduction in diameter from 8.5 to 7.0 ft reduces the heat shield weight to approximately 300 lb for the same design temperature (300°F). An increase in design backface temperature would result in a further reduction in heat shield weight: 270 lb for 400°F structure temperature and 220 lb for 600°F structure;

3) The thermo-structural performance of the heat shield structure unit is of a critical nature because of the high strength and rigidity of the carbon phenolic heat shield.

TABLE 92.- ESA-5500(M) ABLATION PROPERTIES

Property	Value
Plastic density, lb/cu ft	58.0
Char density, lb/cu ft.	30.9
Fraction volatilized.	0.536
Total normal emissivity	0.85
Vapor specific heat, Btu/lb-°R.	0.60
Heat of pyrolysis, Btu/lb	0
Molecular weight of vapors, lb/mole	34
Molecular weight of char oxidation products, lb/mole	28
Molecular weight of gas, lb/mole	40
Heat of vaporization[a] (char), Btu/lb	10 500.
Char recession constants	
Rate limited, A_B, lb/in.^{2} sec atm$^{\frac{1}{2}}$	----
E_R, °R.	----
Oxygen diffusion limited A_R	----
Reaction kinetics	
Preexponential constant, A, sec^{-1}	7.13×10^8
Activation energy, B, °R	34 000.
Reaction order, n	2.0

[a]Char sublimation and nitrogen reactions assume heterogeneous thermochemical equilibrium.

TABLE 93.- SAFETY FACTORS AND CRITERIA

Heating component	Safety factor	
	Stagnation	Cone edge
Radiative		
Nonequilibrium shock.	3.0	3.0
Equilibrium far uv.	3.0	----
Entropy layer	----	2.0
Equilibrium visible and CO (44) . . .	1.5	----
Conical flow.	----	1.5
Convective		
Laminar/turbulent	1.5	1.5
Viscous shear	----	1.5
Criteria		
Laminar/turbulent transistion @ $R_{e_\theta} = 250$		
Ablator backface temp = 300°F @ time of aeroshell jettison		

The heat shield thickness requirements for the three missions are given in table 94. Further heat shield discussion is found in Appendix B.

TABLE 94.- BVS DESIGN HEAT LOADS AND ABLATOR THICKNESS

Entry mode	Stagnation point total heat, Btu/ft^2		Heating time, sec	Ablator material	Ablator thickness, in.		
	Convective	Radiative			Stag point	Cone edge	Base
Flyby	5 830	2 360	17	ESA-5500(M)	.34	.33	.22
Orbital	5 200	1 200	22	ESA-5500(M)	.36	.33	.24
Venus/Mercury swingby	10 200	13 700	----	Carbon-phenolic	.75	.65	.28

Aeroshell Stress Analysis

The conical aeroshell structure has been analyzed. An example of this analysis follows with the entry from orbit mission case. The environment for this case is:

$$V_E = 32\ 000 \text{ fps}$$

$$\gamma_E = -30°$$

$$D = 8.5 \text{ ft}$$

$$\theta = 55° \text{ (cone half angle)}$$

$$R_N = 1.0 \text{ ft}$$

The aeroshell structure consists of five major components:

1) Nose cap;
2) Conical shell or shells;
3) Aft stabilization frame;
4) Payload frame;
5) Afterbody.

Nose cap.- The nose cap is designed as a monocoque shell with a constant radius of curvature. The expression for determinating the required thickness of this monocoque shell in terms of external pressure and radius of curvature is:

$$t = \left[\frac{PR^2}{0.35E}\right]^{\frac{1}{2}} \tag{32}$$

where:

t = required thickness,

P = external pressure,

R = radius of curvature,

E = modulus of elasticity.

This equation applies for $4 < \lambda < 24$ where:

$$\lambda = \left[12\left(1-\mu^2\right)\right]^{\frac{1}{4}} Ro/\left[Rt\right]^{\frac{1}{2}} \tag{33}$$

and

μ = Poisson's ratio,

Ro = base radius of nose cap.

Conical shell structure (and afterbody).- Two methods of construction (sandwich and frame stabilized monocoque) were considered for the shell structure of the aeroshell. Both methods assume a uniform external pressure on a conical shell simply supported at the boundaries. The basic equation for general instability of the shell structure is based on experimental work on homogenous truncated cones subjected to external collapsing pressure. The expression for the general instability allowable is:

$$p = 0.736E/\left(\frac{L}{R}\left(\frac{R}{t}\right)^{5/2}\right) \tag{34}$$

where

p = allowable pressure, psi,

E = modulus of elasticity, psi,

L = slant length of the cone, in.,

R = average slant radius of the cone, in.,

t = thickness of cone skin, in.

The avove equation, modified as required, and additional equations required for local instability checks were used to analyze the conical shells as discussed below.

Sandwich shells.- The sandwich shells are analyzed as two truncated cones because of the location of the payload frame. The analysis assumes both cones to be hydrostatically loaded, i.e., the cones are subjected to a compressive longitudinal stress. The analysis assumes that structures having equal radii of gyration, in the circumferential direction, will work to the same stress level before becoming unstable. By equating radius of gyration,

$$\left(\frac{bt_m^3}{12bt_m}\right)^{\frac{1}{2}} = \left(\frac{2bt_s d^2}{2bt_s}\right)^{\frac{1}{2}} \tag{35}$$

solving for t_m

$$t_m = \sqrt{12d} \tag{36}$$

and using the assumption of equal stress

$$\frac{p_m R}{t_m} = \frac{p_s R}{2t_s} \tag{37}$$

$$p_s = \frac{2t_s p_m}{t_m} \tag{38}$$

the general instability equation becomes:

$$p = \frac{9.49\ E\ t_s d^{3/2}}{LR^{3/2}} \tag{39}$$

where:

t_m = monocoque thickness, in.,

b = unit width, in.,

t_s = sandwich face thickness, in.,

d = distance from sandwich centroid to centroid of face sheet, in.

It is assumed that core cell size is small enough to preclude intercellular buckling of the face sheets.

The above equation has been incorporated in a program written for the IBM 1130 computer that will determine optimum weight sandwich shells within the limits of minimum face thickness and minimum and/or maximum core height. Included are weight for the required core, bonding agent between face sheets and core, and appropriate edge members to allow manufacturing of a segmented aeroshell.

Frame stabilized monocoque shells.- Frame stabilized monocoque shells were analyzed as two shells because of the location of the payload frame. Shells forward of the payload frame were considered hydrostatically loaded and shells aft of the payload frame considered radially loaded by a uniform external pressure. The analysis assumes that structures having equal radii of gyration, in the circumferential direction, will work to the same stress level before becoming unstable. With this assumption the general instability equation becomes:

$$p = \frac{0.736\ E\ \bar{t}}{LR^{3/2}} \left[\frac{12\ I}{b\bar{t}} \right]^{3/4} \tag{40}$$

where:

$\bar{t}$ = smear thickness of structure over a width b, in.,

I = moment of inertia of the structure over a width b and about an axis parallel to the shell skin, in.^{4},

b = frame spacing at the average radius of the cone, in.

In addition to a general instability check, the following checks are made on the detail shell structure.

1) Local instability of intermediate frame elements;

2) Local instability of the shell;

3) Local yield of any shell element.

The local instability check for the intermediate frame elements is expressed in general form as:

$$p = \frac{KR\ \pi^2\ E}{12(1-\mu^2)} \left(\frac{t_r}{b_r} \right)^2 \left[\frac{\bar{t}}{R} \right] \tag{41}$$

where:

I = a coefficient that is a function of the frame element boundary conditions,

R = local radius of curvature of the shell, in.,

t_r = frame element thickness, in.,

b_r = frame element width, in.

Local instability of the cone skin, between frames, is checked by the use of two expressions. The first assumes the skin to be an infinitely long flat panel simply supported at the edges. The second assumes the skin to be a truncated homogenous cone.

The higher allowable of these two expressions is used:

$$1)\quad p = \frac{CE\bar{t}}{R} \left[\frac{t_s}{b} \right]^2 \qquad 2)\quad p = \frac{0.736\ E\bar{t}\ t_s^{\ 3/2}}{bR^{3/2}} \tag{42}$$

where:

C = a coefficient that is a function of the skin element boundary conditions,

t_s = skin thickness,

$\bar{t}$ = smear thickness,

R = local radius of curvature,

b = frame spacing.

The analysis consists of selecting the appropriate element sizes so that all the stability checks are satisfied, the structure will not be critical in yield, and a minimum weight structure will be achieved. This method of analysis was suggested by and has been compared to those proposed in references 20 and 21 and shows very good agreement.

A program for the IBM 1130 computer has been written to perform the analysis. The program handles structure with rectangular-integral frames or channel frames. Variables input to the program include cone angle, minimum gages and dimensions on all elements, design pressure, material allowables, etc. Output consists of detail dimensions, allowables, and weights.

Aft stabilization frame.- The shell analysis considers general instability failure through multiwave buckling of the shell with the ends of the cone assumed to be supported. An end or edge frame is required to provide this support and to prevent general instability in the N = 2 mode of buckling. Because of the restraint provided by the basic shell, the frame will be forced to buckle out-of-plane. The frame moment of inertia required to prevent this buckling is given by the expression:

$$I = \frac{a^4 P \tan^2 \alpha \sin^2 \alpha}{9E} \tag{43}$$

where:

a = slant length of cone extended to the apex, in.,

P = load on frame, lb/in.,

α = 1/2 the included angle of the cone.

A program for the IBM 1130 computer has been written to size a minimum weight frame that will satisfy general and local instability requirements.

Payload frame.- The payload frame consists of structure to redistribute the payload weight during entry into a fairly uniform load into the shell structure of the aeroshell. The payload frame was analyzed as a circular stable web beam on the elastic foundation provided by the aeroshell structure. The method of analysis assumes a straight beam equal in length to the circumference of the payload frame and matched end conditions on either end of the frame. The analysis ignored the stiffness provided by hoop continuity of the frame and, therefore, gave adequate but conservative results. A program has been written for the IBM 1130 computer to size a minimum weight frame that will satisfy stiffness, stability and strength requirements.

Analytical results.- The aeroshell was designed by dynamic pressure during entry. The peak dynamic pressure (24.6 psi limit at the stagnation point) was combined with a structural temperature of 300°F. A safety factor of 1.25 was used on the pressure and any safety factor on temperature was built into the defined ablator thickness.

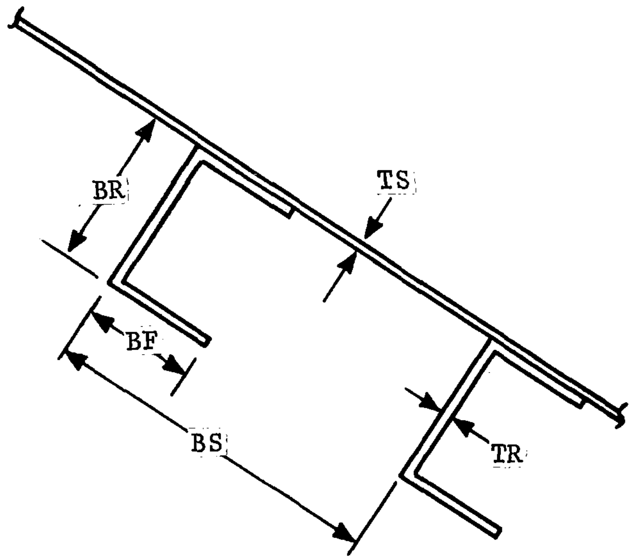
Results of the analysis are shown on figures 135 and 136. Each dimension or gage is defined or restricted by what was considered to be a practical minimum as shown on figure 135. Frame geometries and skin thickness (TS) are constant over each subassembly with the frame spacing varied to optimize the shell.

The aft frame geometry is shown on figure 136. The nose cap was designed as a monocoque shell with a required thickness of 0.035 inches.

BVS Gondola

Figure 137 shows the gondola structure, for the orbital mission, which is a continuation of the semimonocoque structure of the gas inflation system structure. The gondola has two floors on which all the equipment is mounted. The lower floor contains mainly science instruments, while the upper floor contains all the support electronics and antenna. The outer shell as well as the top cover plate are covered with solar cells.

Figure 138 shows the flyby configuration. The difference between the flyby and orbital mission gondolas are as follows:



	Aeroshell			
	Fwd cone	Aft cone	Afterbody	Input minimums
BSF[a]	4.85	2.28	3.5	2.0
BS	2.0	2.0	2.0	2.0
BSA[b]	2.0	2.0	2.0	2.0
TS	.056	.056	.02	.020
BR	.52	.84	.50	.50
BF	.25	.275	.25	.25
TR	.20	.026	.015	.015
Minimum allowable, PSL	30.75	30.75	2.6	

[a] Frame spacing at small end of cone.

[b] Frame spacing at large end of cone.

Figure 135.- Aeroshell and Afterbody Structural Details

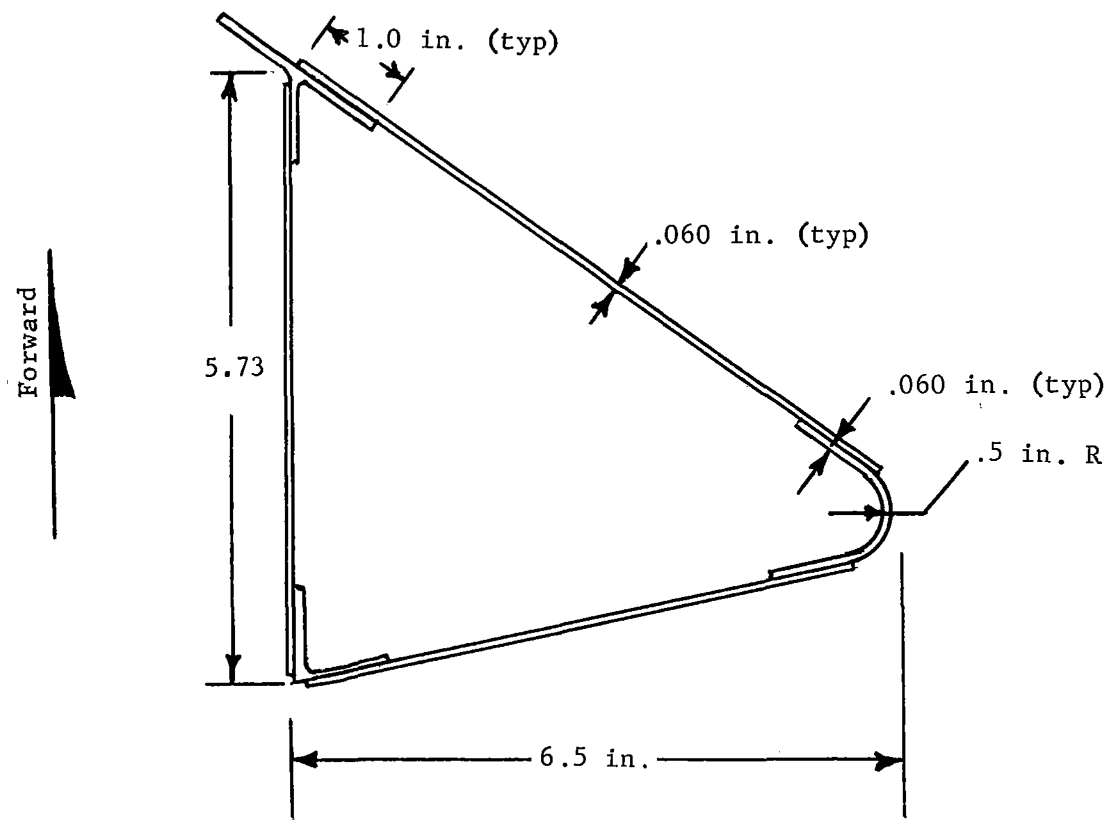


Figure 136.- Aft Stabilization Frame Structural Details

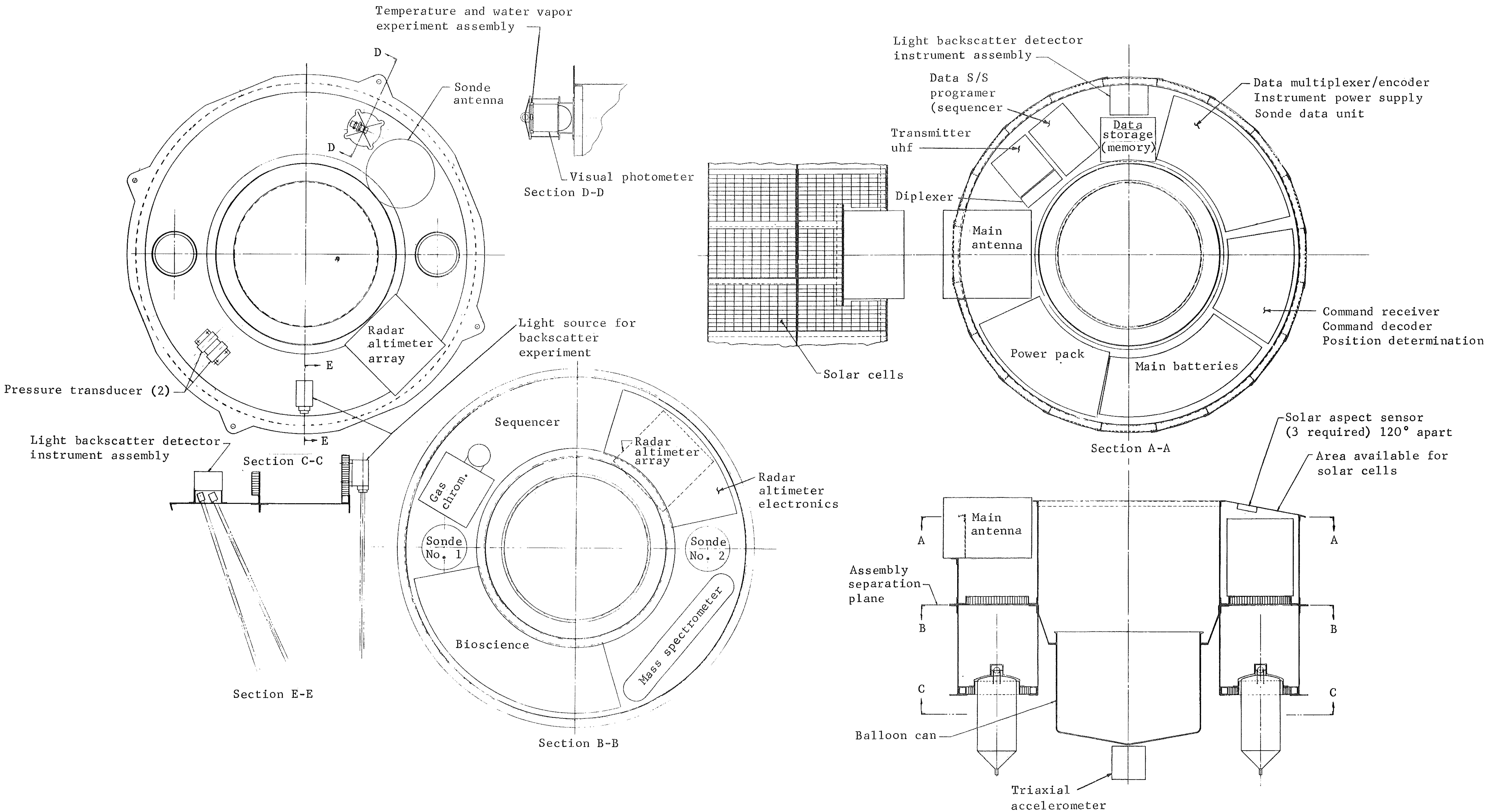


Figure 137.- Gondola Structure and Equipment Layout

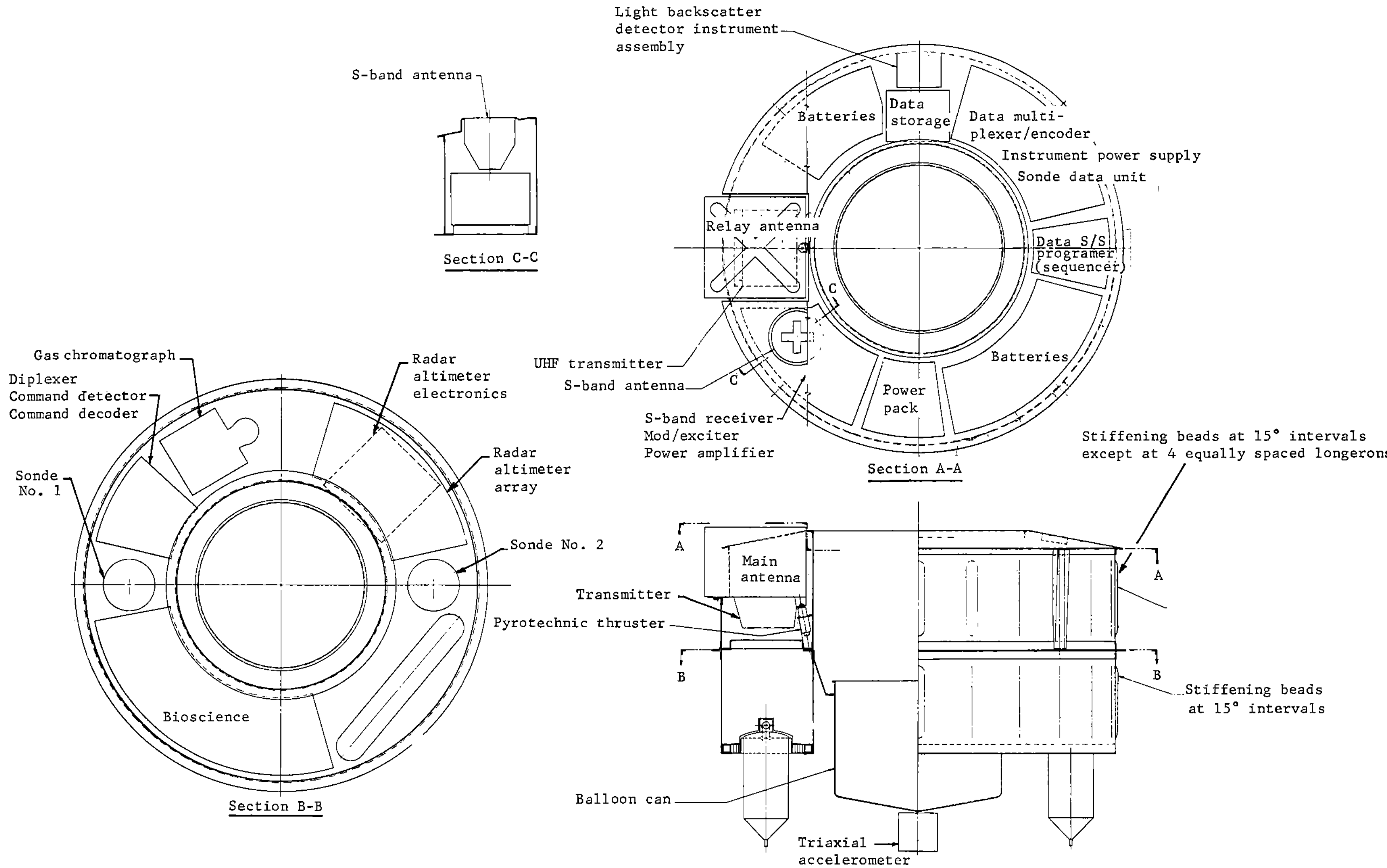


Figure 138.- Gondola Structure and Equipment Layout (Flyby Mission)

1) The cross slot antenna can be jettisoned because it ceases to function at the time the gas inflation system is dropped;

2) An S-band antenna has been added to provide the direct link communication to earth;

3) The solar cells have been removed and replaced by additional batteries. With the removal of solar cells, the stiffeners that served as cell support and skin stiffeners were removed and the skin beaded to compensate for this removal;

4) Miscellaneous equipment changes were made that are related to the addition of the S-band antenna.

The Venus/Mercury gondola is identical to that for the flyby mission shown in figure 138, except for the removal of the jettisonable cross slot antenna and transmitter and associated mechanism hardware. Thus, only the S-band antenna is installed in this gondola.

Gas Inflation Module

Figure 139 shows the gas inflation system and related structure. The structure consists of a straight cylinder structure with ring frames. The four hydrogen tanks are supported off the side by individual tubular members. The upper tension member load will be reacted by the upper ring frame and the lower gondola floor. The lower compression member axial loads will be reacted by the lower ring frame and subsonic probe support cone structure. The lower compression member longitudinal loads will be reacted through a longeron located in the aeroshell cylindrical structure. The tank boss and support provides tangential and longitudinal support at the structure skin.

The four inflation tanks are used to transport ambient temperature, hydrogen gas at a minimal pressure of 4500 psia. The tanks are oblate spheriod in shape, consisting of an 8-mil-thick aluminum liner overwrapped with glass fiber impregnated with epoxy resin. The tank major and minor axes are designed to a ratio of 1.6:1 which is about optimum for a fiber wrapped tank.

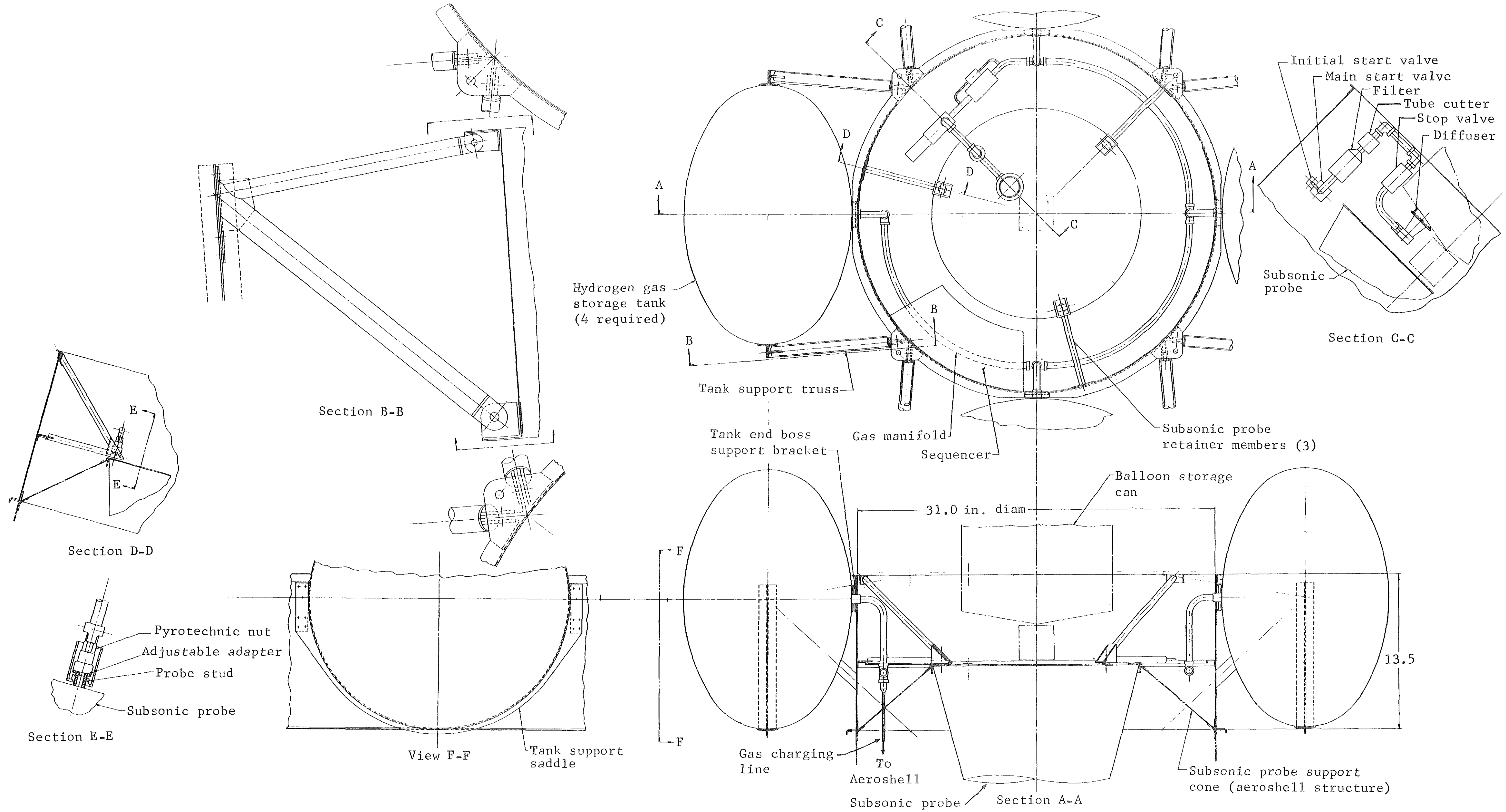


Figure 139.- Gas Inflation System and Structure

The inflation tubing is 0.5 in. in diameter with Parker Weld-Lok fittings shown. These fittings could be changed to Aeroquip Spacecraft Brazing Fittings for easier change of tanks and components.

The subsonic probe is supported by the support cone that is part of the aeroshell structure. When the aeroshell is dropped, the probe is retained with the gas inflation system by three retainer members that hold the release bolts of the probe (see Section E-E, fig. 139 for the attachment details). The reason for making the support cone part of the aeroshell structure and also placing the hydrogen tank longerons in the aeroshell structure is to eliminate weight from the gas inflation system structure, which must be supported by the parachute system.

Balloon/System Interface

Figure 140 shows the configuration of the lower balloon area and balloon can. The inflation diffuser sock is the load carrying member from the gondola through the balloon to the parachute prior to balloon inflation. It also serves to diffuse the gas flow into the balloon. The diffuser sock is designed as a sealed tube from the balloon can to a point near the top of the balloon, forcing the gas to the top of the balloon, filling the top first. After balloon inflation the load is carried into the balloon via the support cone, relieving the inflation sock of any load. For further details of the balloon, refer to the balloon section of this report.

Figure 141 shows the top of the balloon and how it is attached to the parachute system. The pyrotechnic cutter separates the parachute from the balloon just prior to the completion of the balloon fill cycle.

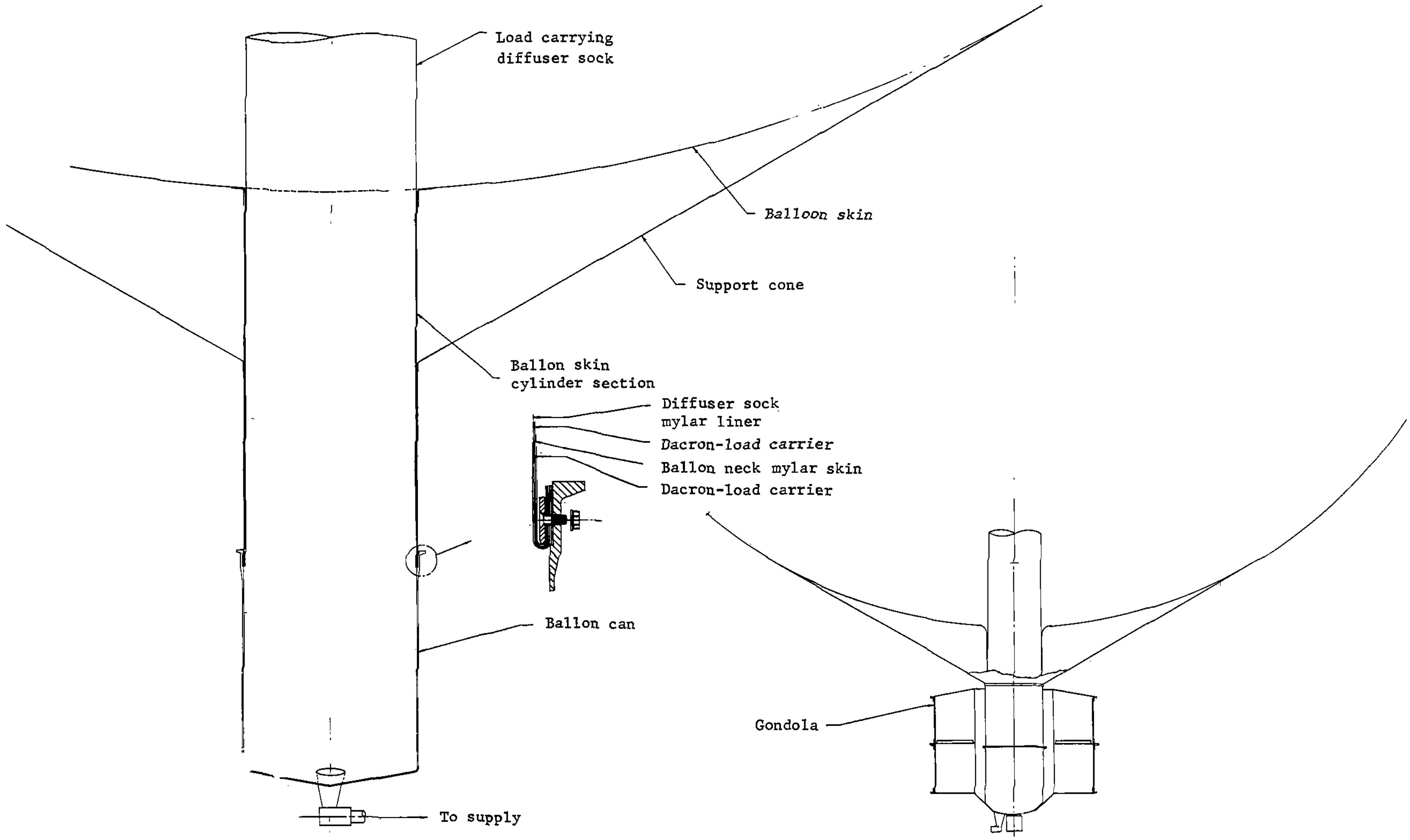


Figure 140.- Ballon Base Gas Diffuser Layout

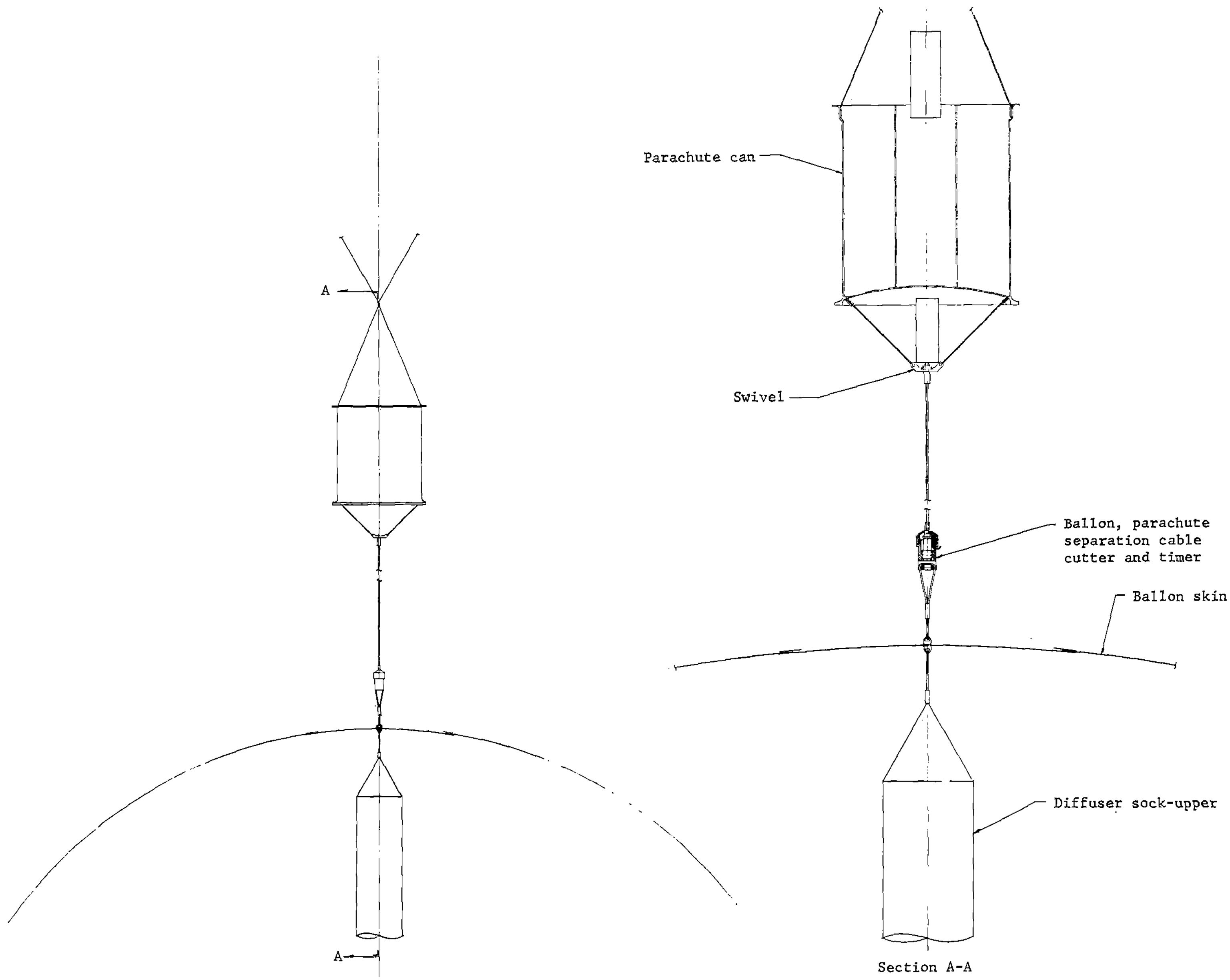


Figure 141.- Ballon to Parachute Can Configuration

WEIGHT AND BALANCE

A most significant measure of feasibility is the weight allocated to the system to perform the mission as compared to the payload weight available. The allocations for the three mission modes considered are summarized in table 95, indicating the overall margins allowed.

Sequential weight statements for the three missions are given in tables 96, 97, and 98. Weight summaries for the BVS are given in tables 99, 100, and 101, and detail weight statements are given in tables 102, 103, and 104.

BVS/EV center of gravity and moment of inertia data are presented in table 105 for various conditions throughout the coast, entry, deployment, and flotation phases of the orbital mission. The reference axis system is shown in figure 142.

The small amount of center of gravity offset, which is critical for a spin stabilized configuration, may be easily corrected. The entry requirement that the roll inertia be a minimum of 10 percent greater than either the pitch or yaw inertias has been adequately met.

TABLE 95.- MISSION WEIGHT ALLOCATIONS SUMMARY

Item	Mission		
	1972 flyby	1973 orbital	1973 Venus/Mercury swingby
Available for planetary vehicle	2426	3675	2496
Allocated for planetary vehicle	2125	3324	2361
Mission margin	301	352	135
Spacecraft in transit	951	2298	1115
Spacecraft at Venus	882	[a]959	1049
Total capsule system	1174	1126	1138
Capsule entry weight	963	827	999
BVS	405	403	412
Subsonic probe	85	85	85
Aeroshell and mounted equipment	473	339	502

[a]The modified lunar orbiter, in orbit, has 661 lb of useful weight plus a 298-lb propulsion module, including residuals.

TABLE 96.- FLYBY MISSION SUMMARY WEIGHT STATEMENT

Item	Weight, lb		
Booster capability			(2600)
Total margin			300.6
Gross payload			(2299.4)
Payload booster adapter			174
Weight to trans-Venus orbit			(2125.4)
Midcourse propellant 75 m/sec			59
ACS propellant			10
Weight to planet			(2056.4)
Spacecraft burnout			882
Structure		183	
Radio		48	
Flight command		8	
Power		109	
Central computor and sequence		24	
Telemetry		24	
Attitude control		72	
Pyrotechnics		11	
Cabling		78	
Thermal control		31	
Mechanical devices		54	
Data storage		50	
Data automation		13	
Scan control		12	
Capsule communications		16	
Science		98	
Propulsion		51	
Total flight capsule			(1174)
Adapter and canister			153
Structure		116	
Insulation		27	
Biofilter		3	
Electrical		7	
Separation weight			(1021.4)
Deflection module		32.2	
Engine	2.7		
Engine mount	2.0		
Fuel tanks (2)	2.7		

TABLE 96.- FLYBY MISSION SUMMARY WEIGHT STATEMENT - Continued

Item	Weight, lb		
Pressure tanks (2)	1.3		
Tank supports	6.6		
Valves and plumbing	10.4		
Electrical (incl separation disconnects)	2.0		
Base structure (incl separation bolts)	3.5		
Residuals	1.0		
Deflection propellant		22.4	
Spin up (solid propellant)		1.0	
Despin yo-yos		3.0	
Entry weight			(962.8)
Afterbody			80.0
Base structure (mag skin-str)		17	
Ablator		58	
rf window		5	
Afterbody chute (10 ft diam)			3.0
Aeroshell			302
Structure	125		
Ablator	177		
Spin system fixed			2.0
Despin system fixed			2.0
Nutation damper			2.0
Equipment in aeroshell			55.5
Power		14.0	
Battery (36.7 W-hr)	8.4		
Transfer switch	1.0		
Power control	1.5		
Isolation diodes	.4		
Wire and connectors	1.5		
Supports	1.2		
Telemetry		8.5	
Engineering sensors	2.0		
Instrument power supply	.5		
Multiplexer encoder	3.0		
Cabling	2.5		
Supports	.5		
Sequencer		6.0	
Sequence package	4.0		
Cables and supports	2.0		
Pyrotechnics			
Control package	12.9		
Cabling	9.6		
Supports	.5		

TABLE 96.- FLYBY MISSION SUMMARY WEIGHT STATEMENT - Concluded

Item	Weight, lb
Structure	16.0
Weight on main chute	(500.3)
Subsonic probe	85
Weight on chute less probe	(415.3)
Balloon can lid	7.9
Main chute and can (32 ft diam)	18.2
Initial weight on balloon	(389.2)
Structure (inflation truss)	21
Hydrogen tanks	128
Residual and valved H_2	1.0
Valves and piping	16.0
Equipment	11.0
uhf	10.0
BVS weight at equilibrium floatation	(202.2)
Structure	16.2
Balloon system	29.3
Power system	26.9
Telecommunications	53.1
Pyrotechnic control	1.6
Science	63.3
Cabling	10.8
Thermal control	1.0

TABLE 97.- ORBITAL MISSION SUMMARY WEIGHT STATEMENT

Item	Weight, lb		
Booster capability			(3850)
Total margin			352.4
Gross payload			3497.6
Payload booster adapter			174
Weight to trans-Venus orbit			
Midcourse and orbit insertion			1329
ACS propellant			10
Weight in Venus orbit			2084.6
Spacecraft burnout			959
Power		138	
Telecommunications		72	
Programer CC&S		32	
Attitude control		62	
Structures and mechanisms		170	
Assembly and integration		78	
Propulsion		298	
Science		109	
Total flight capsule			(1125.6)
Adapter and canister			153
Structure	116		
Insulation	27		
Biofilter	3		
Electrical	6		
Separation weight			(972.6)
Deorbit module			61
Engine (lunar orbiter)		5.5	
Engine mount		2.0	
Fuel tanks (2)		2.8	
Oxidizer tanks (2)		4.5	
Pressure tanks (2)		2.2	
Tank supports		16.0	
Valves and plumbing		16.8	
Electrical		2.0	
Base structure		3.5	
Residuals		5.7	
Deorbit propellant			80.4
Spin up (solid propellant)			1.0
Despin yo-yos			3.0
Entry weight			(827.2)
Afterbody			31.0
Base structure (mag skin-str)		17	
Ablator		9	
rf window		5	

TABLE 97.- ORBITAL MISSION SUMMARY WEIGHT STATEMENT - Continued

Item	Weight, lb		
Afterbody chute (10 ft diam)			3.0
Aeroshell			230
Structure	120		
Ablator	110		
Spin system fixed			2.0
Despin system fixed			2.0
Nutation damper			2.0
Equipment in aeroshell			53.0
Power		8.5	
Battery (36.7 W-hr)	2.6		
Transfer switch	1.0		
Power control	1.5		
Isolation diodes	1.0		
Wire and connectors	1.5		
Supports	0.9		
Telemetry		8.5	
Engineering sensors	2.0		
Instrument power supply	0.5		
Multiplexer encoder	3.0		
Cabling	2.5		
Supports	0.5		
Sequencer		7.0	
Sequence package	5.0		
Cables and supports	2.0		
Pyrotechnics		23.0	
Control package	12.9		
Cabling	9.6		
Supports	0.5		
Science		6.0	
ir photometers (3)	2.0		
uv photometers (3)	3.0		
Supports and cable	1.0		
Structure			16.0
Weight on main chute			(488.2)
Subsonic probe			85
Weight on chute less probe			(403.2)
Balloon can lid			7.9
Main chute and can (32 ft diam)			18.2

TABLE 97.- ORBITAL MISSION SUMMARY WEIGHT STATEMENT - Concluded

Item	Weight, lb
Initial weight on balloon	(377.1)
Structure	21
Hydrogen tanks	128
Residual and valved H_2	1.0
Valves and piping	16.0
Equipment	11.0
BVS weight at equilibrium flotation	(200.1)
Structure	16.7
Balloon system	29.3
Power system	32.8
Telecommunications	45.6
Pyrotechnic control	1.6
Science	63.3
Cabling	10.8

TABLE 98.- VENUS/MERCURY SWINGBY SUMMARY WEIGHT STATEMENT

Item	Weight, lb		
Booster capability			(2670.0)
Total margin			134.5
Gross payload			(2535.5)
Payload booster adaptor			174
Weight to trans-Venus orbit			(2361.5)
Midcourse propellant, 75 m/sec			56
ACS propellant			10
Weight to planet			(2295.5)
Spacecraft burnout			1049
Total flight capsule			(1137.5)
Adapter and canister			113
Structure		83	
Insulation		20	
Biofilter		3	
Electrical		7	
Separation weight			(1133.5)
Deflection module			36.7
Engine	2.7		
Engine mount	2.0		
Fuel tanks (2)	3.5		
Pressure tanks (2)	2.1		
Tank supports	8.5		
Valves and plumbing	9.3		
Electrical (incl separation disconnects)	2.0		
Base structure (incl separation bolts)	3.5		
Residuals	3.1		
Deflection propellant (93 m/sec ΔV)			46.6
ACS system			16.0
Despin yo-yos			3.0
Control package			32.0
Entry weight			(999.2)
Afterbody			75
Base structure (mag skin-str)		30	
Ablator		40	
rf window		5	
Afterbody chute (10 ft diam)			3.0
Aeroshell (7 ft diam)			350
Structure		120	
Ablator		230	

TABLE 98.- VENUS/MERCURY SWINGBY SUMMARY WEIGHT STATEMENT - Concluded

Item	Weight, lb		
Despin system fixed			2.0
Nutation damper			2.0
Equipment in aeroshell			54.4
Power		16.9	
Battery (W-hr)	11.3		
Transfer switch	1.0		
Power control	1.5		
Isolation diodes	.4		
Wire and connectors	1.5		
Supports	1.2		
Telemetry		8.5	
Engineering sensors	2.0		
Instrument power supply	.5		
Multiplexer encoder	3.0		
Cabling	2.5		
Supports	.5		
Sequencer		6.0	
Sequence package	4.0		
Cables and supports	2.0		
Pyrotechnics		23.0	
Control package	12.9		
Cabling	9.6		
Supports	.5		
Structure			16.0
Weight on main chute			(496.8)
Subsonic probe			85
Weight on chute less probe			(411.8)
Balloon can lid			7.9
Main chute and can (32 ft diam)			18.2
Initial weight on balloon			(385.7)
Structure (inflation truss)			21
Hydrogen tanks			128
Residual and valved H_2			1.0
Valves and piping			16.0
Equipment			11.0
BVS weight at equilibrium flotation			(208.7)
Structure			16.2
Balloon system			29.3
Power system			34.4
Telecommunications			52.1
Pyrotechnic control			1.6
Science			63.3
Cabling			10.8
Thermal control			1.0

TABLE 99.- BVS WEIGHT SUMMARY FOR FLYBY MISSION

Item	Weight, lb	
Gondola and contents		177.1
Structure	16.2	
Balloon canister and controls	4.2	
Power	26.9	
Telecommunications	53.1	
Pyrotechnic control	1.6	
Science	58.0	
Science support	5.3	
Cabling	10.8	
Thermal control	1.0	
Balloon and gas		25.1
Balloon	15.6	
Gas	9.5	
Inflation module		177.0
Structure	21.0	
Hydrogen tanks	128.0	
Residual and valved gas	1.0	
Valves and piping	16.0	
Equipment	11.0	
Parachute and miscellaneous		26.1
Balloon can lid	7.9	
Main chute and container	18.2	
Total BVS		405.3

TABLE 100.- BVS WEIGHT SUMMARY FOR ORBITAL MISSION

Item	Weight, lb	
Gondola and contents		175
Structure	16.2	
Balloon canister and controls	4.2	
Power	32.8	
Telecommunications	45.6	
Pyrotechnic control	1.6	
Science	58.0	
Science support	5.3	
Cabling	10.8	
Margin	0.5	
Balloon and gas		25.1
Balloon	15.6	
Gas	9.5	
Inflation module		177.0
Structure	21.0	
Hydrogen tanks	128.0	
Residual and valved gas	1.0	
Valves and piping	16.0	
Equipment	11.0	
Parachute and miscellaneous		26.1
Balloon can lid	7.9	
Main chute and container	18.2	
Total BVS		403.2

TABLE 101.- BVS WEIGHT SUMMARY FOR VENUS/MERCURY SWINGBY MISSION

Item	Weight, lb	
Gondola and contents		183.6
Structure	16.2	
Balloon canister and controls	4.2	
Power	34.4	
Telecommunications	52.1	
Pyrotechnic control	1.6	
Science	58.0	
Science support	5.3	
Cabling	10.8	
Thermal control	1.0	
Balloon and gas		25.1
Balloon	15.6	
Gas	9.5	
Inflation module		177.0
Structure	21.0	
Hydrogen tanks	128.0	
Residual and valved gas	1.0	
Valves and piping	16.0	
Equipment	11.0	
Parachute and miscellaneous		26.1
Balloon can lid	7.9	
Main chute and container	18.2	
Total BVS		411.8

TABLE 102.- BVS WEIGHT DERIVATION FOR FLYBY MISSION

Item	Weight, lb		
Gondola structure			
Upper floor		1.6	
Magnesium skin stringers based on 0.020-in. skin, 3.85 sq ft at 0.6 lb/sq ft			
Lower floor		1.6	
Outer skin and stiffeners		3.2	
Magnesium skin 0.020 in., 1700 sq in. =	2.2		
Magnesium stiffeners 0.020-in. channels 16 in. long, 20 at 0.05 ea =	1.0		
Inner skin		1.3	
Magnesium skin 0.020 in., 900 sq in.			
Chord angles, aluminum		3.5	
Upper outer 0.040 sq in., 31 in. diam =	.4		
Mid outer 0.120 sq in., 31 in. diam =	1.2		
Lower outer 0.060 sq in., 31 in. diam =	.6		
Upper inside 0.040 sq in., 15 in. diam =	.2		
Mid inside 0.120 sq in., 15 in. diam =	.6		
Lower inside 0.100 sq in., 15 in. diam =	.5		
Sonde supports, magnesium		.3	
Balloon can, lower portion		3.0	
Magnesium			
Lip extrusion 0.4 sq in., 13½ in. diam =	1.4		
Cylinder 0.032 in., 8½ in. high, 13 in. diam =	.8		
Bottom 0.032 in., 13 in. diam	.3		
Weld and attachment	.5		
Separation bolts, gondola to inflation module, 2 at 0.6 lb		1.2	
Misc hardware		.5	
Total structure			16.2
Balloon system			
Balloon attachment		.5	
Piping, valves		2.0	
Diffuser, stainless steel	.5		
Shutoff valve	1.1		
Piping	.2		
Supports	.2		
Balloon control		1.7	
Pressure switch	0.5		
Solenoid valve	1.0		
Piping	0.2		
Total Balloon System			4.2

TABLE 102.- BVS WEIGHT DERIVATION FOR FLYBY MISSION - Continued

Item	Weight, lb	
Thermal control		
Phase change material for S-band transmitter	1.0	
Total thermal control		1.0
Pyrotechnic control		
Capacitor assembly	.3	
Safe/arm SW	.5	
Squib fire SW	.4	
Packaging and supports	.3	
Total pyrotechnic control		1.6
Science		
Radar alt (elec)	8.0	
Radar alt array	4.5	
Bioscience	10.0	
Triax	1.5	
Pressure	1.0	
Temperature, 2 at 0.65 lb	1.3	
Light backscatter	1.5	
Solar aspect	1.5	
Water vapor	.8	
Visual photometer	2.0	
Gas chromatograph	16.0	
Science supports	4.7	
Science instruments power supply	.5	
Sondes, 2 at 5.0	10.0	
Total science		63.3
Cabling		
Based upon detail estimate of wire runs and connectors using H-film wire with a minimum of No. 26 gage and microdot connectors		
Total cabling		10.8
Total gondola		177.1

TABLE 102.- BVS WEIGHT DERIVATION FOR FLYBY MISSION - Continued

Item	Weight, lb		
Power system			
50 hr silver-zinc, 13 A-hr (based upon 400 W-hr at 30 V)		20.7	
Isolation diodes		.2	
Miscellaneous equipment			
Undervoltage sensor			
Charge control			
Power control	3.0		
Power transfer	1.0		
Package and cable	1.0	5.0	
Supports		1.0	
Total power system			26.9
Telecommunications			
Telemetry			
Multiplexer encoder		10.0	
Memory		4.0	
Transducer power supply		1.0	
Data handling			
Data handling programer		4.0	
Drop sonde system			
Sonde rec and data assembly		5.0	
Sonde antenna		.7	
Command			
Command detector		4.0	
Command decoder		3.0	
S-band			
S-band rec		5.0	
Modulator exciter		3.0	
Power amplifier		7.8	
Antenna		.6	
Diplexer		.5	
Supports		3.5	
UHF ejection mechanism		1.0	
Total telecommunications			53.1

TABLE 102.- BVS WEIGHT DERIVATION FOR FLYBY MISSION - Continued

Item	Weight, lb		
Balloon and gas			
Balloon envelope, 1.3-mil Mylar, 18 ft diam =	10.6		
Miscellaneous parts =	5.0		
Total baloon		15.6	
Gas in baloon (hydrogen) 3037 cu ft at 618.4 millibars and 300°K		9.5	
Total balloon and gas			25.1
Inflation module structure			
Shell, aluminum monocoque waffle 31 in. diam, 13.5 in. high, 0.052 equivalent thickness		7.5	
Longerons, 4 at 0.2 lb		.8	
Upper and lower frames, aluminum		2.0	
Separation bolts, 4 at 0.6 lb		2.4	
Tank support trusses			
Upper tubes, 3/4 x 0.035-in. Ti, 8 at 0.1 lb		.8	
Diagonal tubes, 1-1/8 x 0.035 Ti, 8 at 0.25 lb		2.0	
Tube end fittings, titanium		3.0	
Center fitting and channel, 4 at 0.4 lb		1.6	
Miscellaneous		.9	
Total structure			21.0
Hydrogen tanks			
Glass wound over aluminum or stainless steel liners			
Glass density 0.09 lb/cu in.			
Impregnated 70% with resin of density 0.07 to 0.08 lb/cu in.			
Thus, net density 0.075 lb/cu in.			
Glass filament strength 480 000 psi			
Translation efficiency, 80%			
Cylinder directional factor, 66.6%			
Thus equivalent strengths 480 000 x 0.7 x 0.8 x 0.666 = 180 000 psi			

TABLE 102.- BVS WEIGHT DERIVATION FOR FLYBY MISSION - Continued

Item		Weight, lb
Winding weight		
Weight = $\frac{2\ PV\ \rho}{\sigma}$		
P = Pressure x safety factory		
4500 x 2.2 = 9.900 psi		
V = Volume cu in.		
ρ = Net composite density = 0.075 lb/cu in.		
σ = Equipment strength = 180 000 psi		
Weight composite = $\frac{2 \times 9900 \times 3330 \times .075}{180\ 000}$ =	27.5	
Liner	1.8	
Miscellaneous fittings, etc.	2.7	
	32.0	
Total tanks 4 at 32.0 lb		128
Residual and valved gas		
Valved gas 10% of 9.5 =	.95	
Residual	.05	
Total residual and valved gas		1.0
Valves and plumbing		
Piping 5/8 x 0.083-in. steel, 150 inches at 0.04 lb/in.	6.0	
Fittings (5) elbows (3), tees	1.7	
Filter	.9	
Tubing cutter	1.0	
Main start valve	1.1	
Initial start valve	.3	
Fill system components	1.6	
Supports and miscellaneous	3.4	
Total valves and piping		16.0

TABLE 102.- BVS WEIGHT DERIVATION FOR FLYBY MISSION - Concluded

Item	Weight, lb		
Equipment			
Sequencer		3.0	
Pyrocontrol package		6.3	
Pyrocabling		1.2	
Miscellaneous		.5	
Total equipment			11.0
Total inflation module			177.0
Parachute and miscellaneous			
Balloon can lid			
Lip extrusion, magnesium		1.4	
Plate 0.032-in. magnesium		.3	
Clamp and mechanism		4.3	
Clamp	1.5		
Fixed bolts	.5		
Explosive bolts	1.8		
Miscellaneous	.5		
Balloon tie cutter		1.9	
Capacitor	.3		
Safe arm	.5		
Squib fire	.1		
Timer	.5		
Pyrocutter	.3		
Miscellaneous	.2		
Total can lid			7.9
Main Parachute			
Cloth and lines, 32 ft diam	12.0		
Can and mortar	6.2		
Total main chute			18.2
Afterbody parachute			
Cloth and lines loft	2.0		
Can and mortar	1.0		
Total afterbody chute			3.0
Total parachute and miscellaneous			29.1

TABLE 103.- BVS WEIGHT DERIVATION FOR ORBITAL MISSION

Item	Weight, lb		
Gondola structure			
Upper floor		1.6	
Magnesium skin stringers based on 0.020-in. skin, 3.85 sq ft at 0.6 lb/sq ft			
Lower floor same as upper		1.6	
Outer skin and stiffeners		3.2	
Magnesium skin, 0.020 in., 1700 sq in. =	2.2		
Magnesium stiffeners, 0.020-in. channels 16 in. long, 20 at 0.05 ea =	1.0		
Inner skin		1.3	
Magnesium skin, 0.020 in., 900 sq in.			
Chord angles, aluminum		3.5	
Upper outer, 0.040 sq in., 31 in. diam =	.4		
Mid outer, 0.120 sq in., 31 in. diam =	1.2		
Lower outer, 0.060 sq in., 31 in. diam =	.6		
Upper inside, 0.040 sq in., 15 in. diam =	.2		
Mid inside 0.120 sq in., 15 in. diam =	.6		
Lower inside, 0.100 sq in., 15 in. diam =	.5		
Sonde supports, magnesium		.3	
Balloon can, lower portion		3.0	
Magnesium			
Lip extrusion, 0.4 sq in., 13½ in. diam =	1.4		
Cylinder, 0.032 in., 8½ in. high, 13 in. diam =	.8		
Bottom, 0.032 in., 13 in. diam	.3		
Weld and attachment	.5		
Separation bolts, gondola to inflation module, 2 at 0.6 lb.		1.2	
Miscellaneous hardware		0.5	
Total structure			16.2
Balloon system			
Balloon attachment		.5	
Piping, valves		2.0	
Diffuser, stainless steel	.5		
Shutoff valve	1.1		
Piping	.2		
Supports	.5		
Balloon control		1.7	
Pressure switch	0.5		
Solenoid valve	1.0		
Piping	0.2		
Total balloon system			4.2

TABLE 103.- BVS WEIGHT DERIVATION FOR ORBITAL MISSION - Continued

Item	Weight, lb		
Power system			
Solar panels, 11.5 sq ft at 0.5 lb/sq ft =		5.8	
50 hr silver-zinc, 13 A-hr (based on 400 W-hr at 30 V)		15.5	
Isolation diodes		1.0	
Miscellaneous equipment			
Undervoltage sensor	.5		
Charge control	2.5		
Power control	3.0		
Power transfer	1.0		
Auxiliary equipment	1.0		
Package and cable	1.5	9.5	
Total power system			32.8
Telecommunications			
Telemetry			
Multiplexer encoder		10.0	
Memory		4.0	
Transducer power supply		1.0	
Data handling			
Data handling programer		4.0	
Drop sonde system			
Sonde rec. and data assembly		5.0	
Sonde antenna		.7	
Command			
Command receiver		3.0	
Command decoder		1.5	
Position determination data unit (1-way doppler rec.)		3.0	
Transmission			
UHF 20 W transmitter		4.0	
Antenna		5.0	
Diplexer		.6	
Supports		3.0	
Total telecommunications			45.6
Science			
Radar alt (elect.)		8.0	
Radar alt array		4.5	
Bioscience		10.0	
Triax		1.5	
Pressure		1.0	
Temperature, 2 at 0.65 lb		1.3	

TABLE 103.- BVS WEIGHT DERIVATION FOR ORBITAL MISSION - Continued

Item	Weight, lb		
Light backscatter		1.5	
Solar aspect		1.5	
Water vapor		.8	
Visual photometer		2.0	
Gas chromatograph		16.0	
Science supports		4.7	
Science instrument power supply		.5	
Sondes, 2 at 5.0 lb		10.0	
Total science			63.3
Cabling			
Based on detail estimate of wire runs and connectors using H-film wire with a minimum of No. 26 gage and microdot connectors			
Total cabling			10.8
Margin			0.5
Total gondola			175.0
Balloon and gas			
Balloon envelope - 1.3-mil Mylar, 18 ft diam =	10.6		
Miscellaneous parts =	5.0		
Total balloon		15.6	
Gas in balloon (hydrogen) 3037 cu ft at 618.4 mb and 300°K		9.5	
Total balloon and gas			25.1
Inflation module structure			
Shell, aluminum monocoque waffle, 31 in. diam, 13.5 in. high, 0.052-in. equivalent thickness		7.5	
Longerons, 4 at 0.2 lb		.8	
Upper and lower frames, aluminum		2.0	
Separation bolts 4 at 0.6 lb		2.4	
Tank support trusses			
Upper tubes, 3/4 x 0.035 in.-Ti, 8 at 0.1 lb		.8	
Diagonal tubes, 1-1/8 x 0.035-in. Ti, 8 at 0.25 lb		2.0	
Tube end fittings, titanium		3.0	
Center fitting and channel, 4 at 0.4 lb		1.6	
Miscellaneous		.9	
Total structure			21.0

TABLE 103.- BVS WEIGHT DERIVATION FOR ORBITAL MISSION - Continued

Item	Weight, lb	
Hydrogen tanks		
Glass wound over aluminum or stainless steel liners		
Glass density, 0.09 lb/cu in.		
Impregnated 70% with resin of density 0.07 to 0.08 lb/cu in.		
Thus, net density 0.075 lb/cu in.		
Glass filament strength 480 000 psi		
Translation efficiency 80%		
Cylinder directional factor 66.6%		
Thus equivalent strengths 480 000 x 0.7 x 0.8 x 0.666 = 180 000 psi		
Winding weight		
Weight = $\frac{2\ PV\ \rho}{\sigma}$		
P = Pressure x safety factor 4500 x 2.2 = 9.900 psi		
V = Volume cu in.		
ρ = Net composite density = .075 lb/cu in.		
σ = Equipment strength = 234 000 psi		
Weight composite = $\frac{2 \times 9900 \times 3330 \times 0.075}{180\ 000}$ =	27.5	
Liner	1.8	
Miscellaneous fittings, etc	2.7	
	32.0	
Total tanks, 4 at 32.0 lb =		128
Residual and valved gas		
Valved gas, 10% of 9.5 =	.95	
Residual	.05	
Total residual and valved gas		1.0
Valves and plumbing		
Piping, 5/8 x 0.083-in. steel, 150 in. at 0.04 lb/in.	6.0	
Fittings (5), Elbows (3), tees	1.7	
Filter	.9	
Tubing cutter	1.0	
Main start valve	1.1	

TABLE 103.- BVS WEIGHT DERIVATION FOR ORBITAL MISSION - Concluded

Item	Weight, lb		
Initial start valve	.3		
Fill system components	1.6		
Supports and miscellaneous	3.4		
Total valves and piping			16.0
Equipment			
Sequencer		3.0	
Pyrocontrol package		6.3	
Pyrocabling		1.2	
Miscellaneous		.5	
Total equipment			11.0
Total inflation module			177.0
Parachute and miscellaneous			
Balloon can lid			
Lip extrusion, magnesium		1.4	
Plate, 0.032-in., magnesium		.3	
Clamp and mechanism		4.3	
Clamp	1.5		
Fixed bolts	.5		
Explosive bolts	1.8		
Miscellaneous	.5		
Balloon tie cutter		1.9	
Capacitor	.3		
Safe arm	.5		
Squib fire	.1		
Timer	.5		
Pyrocutter	.3		
Miscellaneous	.2		
Total can lid			7.9
Main parachute			
Cloth and lines, 32 ft diam		12.0	
Can and mortar		6.2	
Total main chute			18.2
Afterbody parachute			
Cloth and lines loft		2.0	
Can and mortar		1.0	
Total afterbody chute			3.0
Total parachute and miscellaneous			29.1

TABLE 104.- BVS WEIGHT DERIVATION FOR VENUS/MERCURY SWINGBY MISSION

Item	Weight, lb		
Gondola structure			
Upper floor		1.6	
Magnesium skin stringers based on 0.020-in. skin, 3.85 sq ft at 0.6 lb/sq ft			
Lower floor same as upper		1.6	
Outer skin and stiffeners		3.2	
Magnesium skin, 0.020 in., 1700 sq in. =	2.2		
Magnesium stiffeners, 0.020-in. channels 16 in. long, 20 at 0.05 lb ea =	1.0		
Inner skin		1.3	
Magnesium skin, 0.020 in., 900 sq in.			
Chord angles, aluminum		3.5	
Upper outer, 0.040 sq in., 31 in. diam =	.4		
Mid outer, 0.120 sq in., 31 in. diam =	1.2		
Lower outer, 0.060 sq in., 31 in. diam =	.6		
Upper inside, 0.040 sq in., 15 in. diam =	.2		
Mid inside, 0.120 sq in., 15 in. diam =	.6		
Lower inside, 0.100 sq in., 15 in. diam =	.5		
Sonde supports, magnesium		.3	
Balloon can, lower portion		3.0	
Magnesium			
Lip extrusion, 0.4 sq in., 13½ in. diam =	1.4		
Cylinder, 0.032 in., 8½ in. high, 13 in. diam	.8		
Bottom, 0.032 in., 13 in. diam	.3		
Weld and attachment	.5		
Separation bolts, gondola to inflation module, 2 at 0.6 lb		1.2	
Miscellaneous hardware		.5	
Total structure			16.2
Balloon system			
Balloon attachment		.5	
Piping, valves		2.0	
Diffuser, stainless steel	.5		
Shutoff valve	1.1		
Piping	.2		
Supports	.2		
Balloon control		1.7	
Pressure switch	0.5		
Solenoid valve	1.0		
Piping	0.2		
Total balloon system			4.2

TABLE 104.- BVS WEIGHT DERIVATION FOR VENUS/MERCURY SWINGBY MISSION - Continued

Item	Weight, lb		
Power system			
50-hr silver-zinc battery		28.2	
Isolation diodes		.2	
Miscellaneous equipment			
Power control	3.0		
Power transfer	1.0		
Package and cable	1.0	5.0	
Supports		1.0	
Total power system			34.4
Telecommunications			
Telemetry			
Multiplexer encoder		10.0	
Memory		4.0	
Transducer power supply		1.0	
Data handling			
Data handling programer		4.0	
Drop sonde system			
Sonde rec. and data assembly		5.0	
Sonde antenna		.7	
Command			
Command detector		4.0	
Command decoder		3.0	
S-band			
S-band rec.		5.0	
Modulator exciter		3.0	
Power amplifier		7.8	
Antenna		.6	
Diplexer		.5	
Supports		3.5	
Total telecommunications			52.1
Thermal control			
Phase change material for S-band transmitter		1.0	
Total thermal control			1.0
Pyrotechnic control			
Capacitor assembly		.3	
Safe/arm SW		.5	
Squib fire SW		.4	
Packaging and supports		.3	
Total pyrotechnic control			1.6

TABLE 104.- BVS WEIGHT DERIVATION FOR VENUS/MERCURY SWINGBY MISSION - Continued

Item	Weight, lb		
Science			
Radar alt. (elect.)		8.0	
Radar alt. array		4.5	
Bioscience		10.0	
Triax		1.5	
Pressure		1.0	
Temperature, 2 at 0.65 lb		1.3	
Light backscatter		1.5	
Solar aspect		1.5	
Water vapor		.8	
Visual photometer		2.0	
Gas chromatograph		16.0	
Science supports		4.7	
Science instruments power supply		.5	
Sondes, 2 at 5.0 lb		10.0	
Total science			63.3
Cabling			
Based upon detail estimate of wire runs and connectors using H-film wire with a minimum of No. 26 gage and microdot connectors			
Total cabling			10.8
Total gondola			177.1
Balloon and gas			
Balloon envelope, 1.3-mil Mylar, 18 ft diam =	10.6		
Miscellaneous parts =	5.0		
Total balloon		15.6	
Gas in balloon (hydrogen) 3037 cu ft at 618.4 mb and 300°K		9.5	
Total balloon and gas			25.1
Inflation module structure			
Shell, aluminum monocoque waffle 31 in. diam, 13.5 in. high, 0.052 eq thickness		7.5	
Longerons, 4 at 0.2 lb		.8	
Upper and lower frames, aluminum		2.0	
Separation bolts, 4 at 0.6 lb		2.4	

TABLE 104.- BVS WEIGHT DERIVATION FOR VENUS/MERCURY SWINGBY MISSION - Continued

Item	Weight, lb	
Tank support trusses		
Upper tubes, 3/4 x 0.035-in. Ti, 8 at 0.1 lb	.8	
Diagonal tubes, 1-1/8 x 0.035-in. Ti, 8 at 0.25 lb	2.0	
Tube end fittings, titanium	3.0	
Center fitting and channel, 4 at 0.4 lb	1.6	
Miscellaneous	.9	
Total structure		21.0
Hydrogen tanks		
Glass wound over aluminum or stainless steel liners		
Glass density 0.09 lb/cu in. Impregnated 70% with resin of density 0.07 to 0.08 lb/cu in. Thus, net density 0.075 lb/cu in. Glass filament strength 480 000 psi Translation efficiency 80% Cylinder directional factor 66.6% Thus, equivalent strengths 480 000 x 0.7 x 0.8 x 0.666 = 180 000 psi Winding weight $\text{Weight} = \frac{2\ PV\ \rho}{\sigma}$ P = Pressure x safety factory 4500 x 2.2 = 9.900 psi V = Volume cu in. ρ = Net composite density = 0.075 lb/cu in. σ = Equivalent strength = 180 000 psi		
$\text{Weight composite} = \frac{2 \times 9900 \times 3330 \times .075}{180\ 000} =$	27.5	
Liner	1.8	
Miscellaneous fittings	2.7	
	32.0	
Total tanks, 4 at 32.0 lb =		128
Residual and valved gas		
Valved gas 10% of 9.5 lb =	.95	
Residual	.05	
Total residual and valved gas		1.0

TABLE 104.- BVS WEIGHT DERIVATION FOR VENUS/MERCURY SWINGBY MISSION - Continued

Item	Weight, lb		
Valves and plumbing			
Piping, 5/8 x 0.083-in. steel, 150 in. at 0.04 lb/in.	6.0		
Fittings (5), elbows (3), tees	1.7		
Filter	.9		
Tubing cutter	1.0		
Main start valve	1.1		
Initial start valve	.3		
Fill system components	1.6		
Supports and miscellaneous	3.4		
Total valves and plumbing			16.0
Equipment			
Sequencer		3.0	
Pyrocontrol package		6.3	
Pyrocabling		1.2	
Miscellaneous		.5	
Total equipment			11.0
Total inflation module			177.0
Parachute and miscellaneous			
Balloon can lid			
Lip, extrusion, magnesium		1.4	
Plate, 0.032 in. magnesium		.3	
Clamp and mechanism		4.3	
Clamp	1.5		
Fixed bolts	.5		
Exp bolts	1.8		
Miscellaneous	.5		
Balloon tie cutter		1.9	
Capacitor	.3		
Safe arm	.5		
Squib fire	.1		
Timer	.5		
Pyrocutter	.3		
Miscellaneous	.2		
Total can lid			7.9
Main parachute			
Cloth and lines, 32 ft diam		12.0	
Can and mortar		6.2	
Total main chute			18.2

TABLE 104.- BVS WEIGHT DERIVATION FOR VENUS/MERCURY SWINGBY MISSION - Concluded

Item	Weight, lb	
Afterbody parachute		
Cloth and lines loft	2.0	
Can and mortar	1.0	
Total afterbody chute		3.0
Total parachute and miscellaneous		29.1

TABLE 105.- BVS CENTER OF GRAVITY AND MOMENT OF INERTIA DATA

Condition	Weight, lb	Center of gravity			Inertia, slug-ft^2		
		X	Y [a]	Z	I_{XX}	I_{YY}	I_{ZZ}
Total flight capsule	1126	0.05	0.06	38.1	145.3	144.3	154.3
Separation weight	971	-0.05	0.07	40.0	102.3	100.3	99.1
Deorbit burnout	887	-0.06	0.07	42.3	88.3	86.8	96.6
Entry weight	827	-0.06	0.08	44.2	78.6	77.1	96.2
Initial weight on chute	488	-0.01	0.08	21.0	604.2	603.8	29.7
Weight on chute less probe	403	0.13	0.10	13.6	576.1	575.7	27.6
Initial weight on balloon	376	0.14	0.11	30.5	90.8	90.4	27.4
Balloon floating	200	-0.26	0.43	19.3	70.2	69.9	9.2
Balloon, 1 sonde dropped	195	-0.27	0.75	18.7	69.4	69.3	9.0
Balloon, no sondes	190	-0.28	0.45	18.0	68.6	68.7	8.9

a. Reference figure 142, page 403

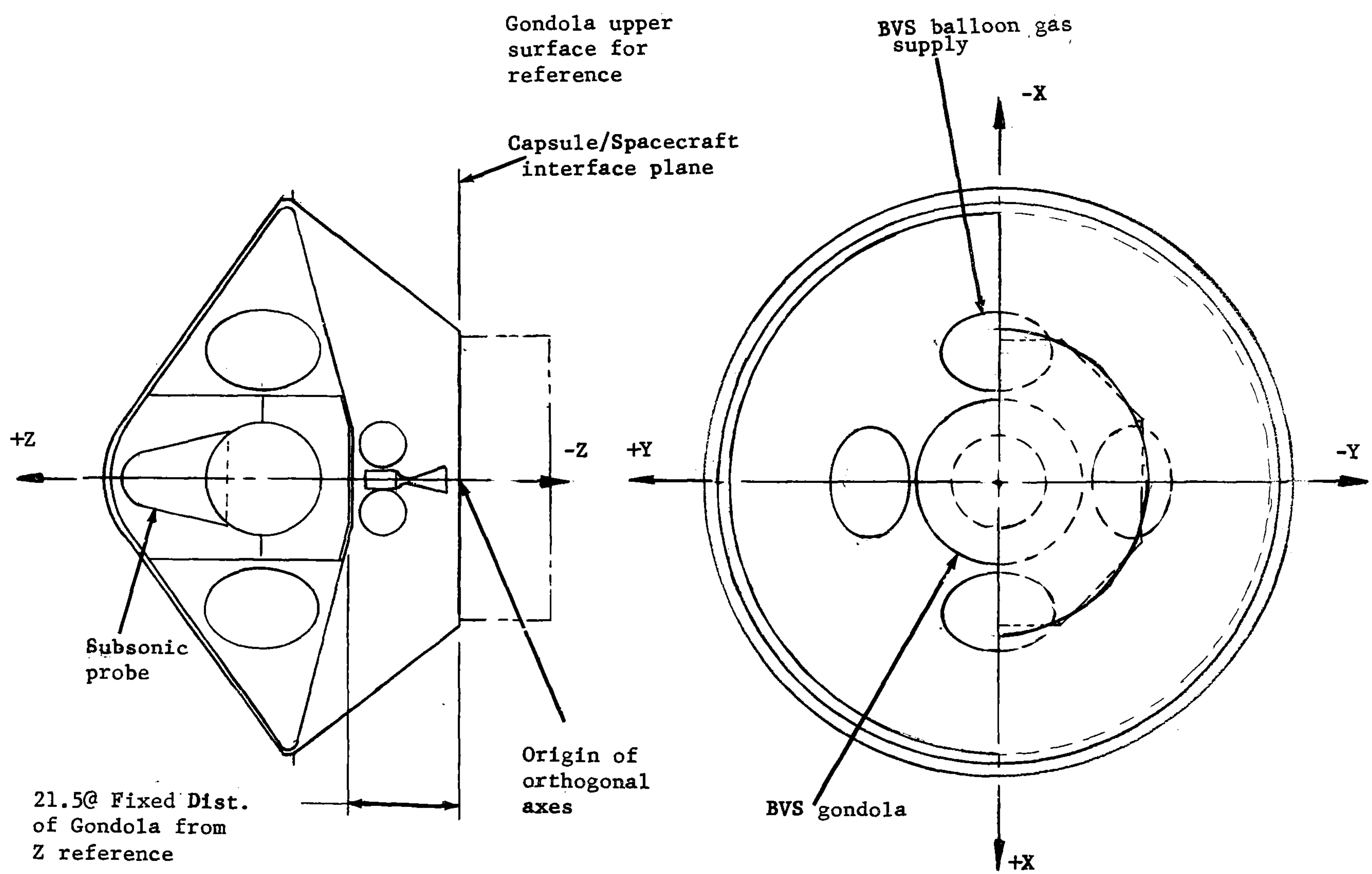


Figure 142. - Planetary Vehicle Coordinate System

The mission sequencing requirements are nearly identical for the three missions. Two sequencers, one in the aeroshell and one in the BVS, are required with a similar number of discrete outputs for the three missions (with different intervals required for the aeroshell sequencer because of the variation in coast period from 10 to 68 hr).

Spin stabilization is used for control during coast for the orbital and flyby missions. For the Venus/Mercury mission an active system is required because of the reorientation required for entry after the deflection maneuver.

Aeroshell Sequencer

The aeroshell sequencer provides discretes to control power transfer, to control operating mode, to control data formats, and to initiate pyro arm, fire, and safe functions from before spacecraft separation to separation of the BVS. The performance requirements are listed in table 106.

TABLE 106.- AEROSHELL SEQUENCER REQUIREMENTS (TYPICAL)

1. Sequencer	
Separation	18 discretes
Preentry	4 discretes
Postentry	8 discretes
2. Maximum timing interval. . .	12 hr
3. Inflight reprograming	
ΔV burntime	100 to 234 sec
Deorbit coast time . . .	8 to 12 hr
4. Timing accuracy	0.1%
5. Resolution	
ΔV burntime	0.1 sec
All others	1.0 sec
6. Discrete duration	Momentary
7. Special requirements	Prelaunch system checkout capability. Inhibit pyro discretes if biocanister or capsule have not separated.

The aeroshell sequencer block diagram is shown in figure 143. The mechanization includes a magnetic oscillator, a clock divider, a timer counter, four discrete matrix detectors, three sets of output switches, two register detectors, an update register, and a command decoder. Before separation the deorbit engine burntime is updated from earth via the spacecraft. The coast timer is also initiated before separation based on the predicted time to entry. The separation sequence is then initiated by the spacecraft.

If the biocanister or the capsule fails to separate, an inhibit is applied to the output that prevents subsequent discretes from being issued, thus preventing damage to the spacecraft. To conserve power, a powerdown mode for coast is initiated after the final discrete in the separation sequence. The sequencer is powered up once again by the coast timer before entry and the preentry sequence is completed.

The postentry sequence is initiated by a signal from the longitudinal accelerator when a 1 g increasing deceleration level is detected. The aeroshell sequencer mission is complete when the BVS is separated from the aeroshell. The checkout capability is not shown, however, it consists of a timer counter, a discrete detector, and a set of output switches. Timing will be from the ground equipment. The weight of the sequencer is estimated to be 5 lb and requires 4 W. During the coast period 1 W will be required.

BVS Sequencer

The BVS sequencer provides discretes to control power transfer, to control operating modes, to control data formats, and to initiate pyro arm, fire, and safe functions from BVS/aeroshell separation to BVS and science deployment. The requirements are listed in table 107.

TABLE 107.- BVS SEQUENCER REQUIREMENTS (TYPICAL)

1.	Sequencer	
	Probe separation	4 discretes
	BVS and science deployment . . .	12 discretes
2.	Maximum timing interval	30 min.
3.	Timing accuracy	0.1%
4.	Discrete duration	1.0 sec
5.	Special requirements	Prelaunch system checkout capability

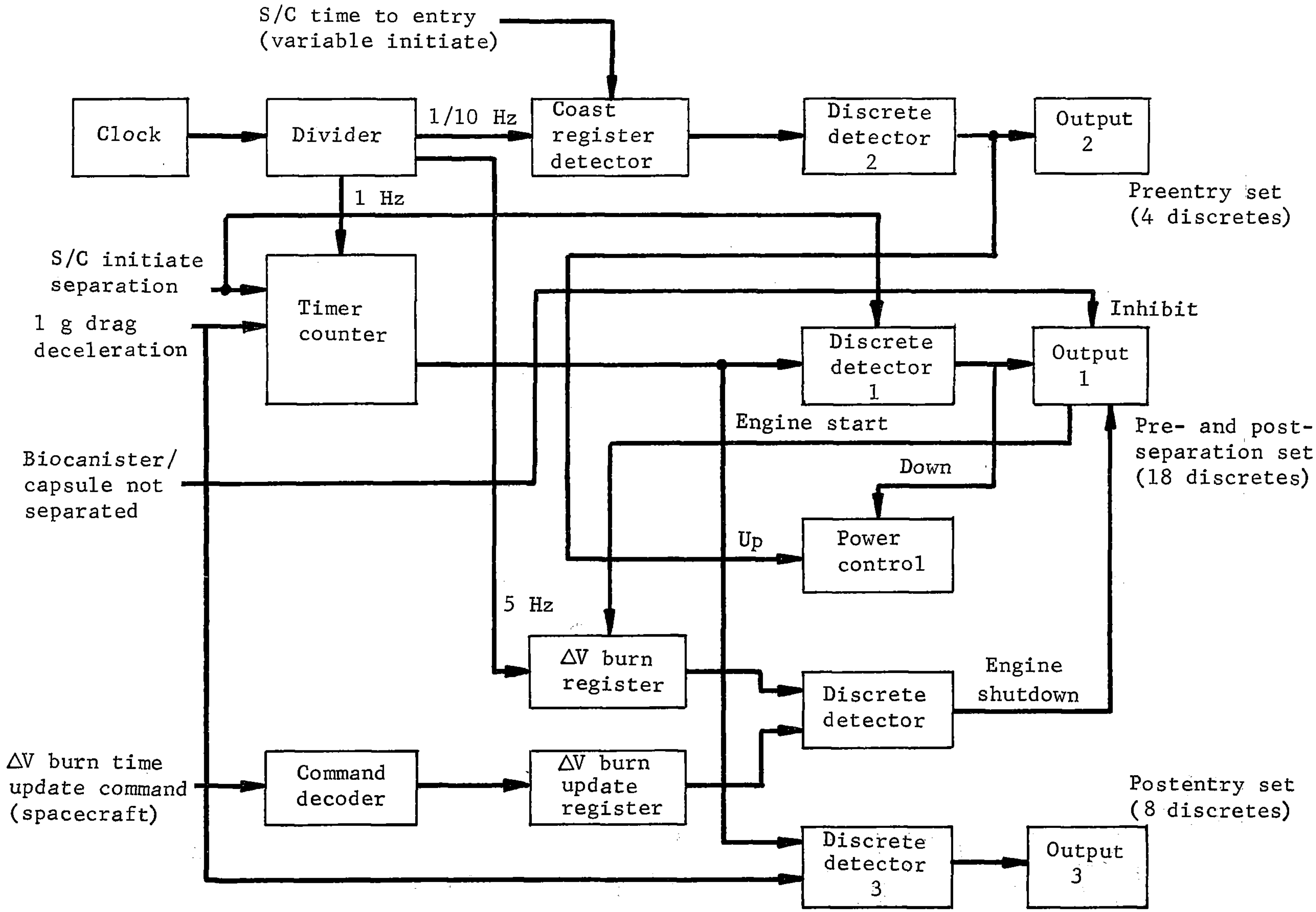


Figure 143.- Aeroshell Sequencer

The BVS sequencer is shown in figure 144. The mechanization includes a magnetic oscillator, a clock divider, a timer counter, two discrete matrix detectors, and two sets of output switches. Power is applied to the sequencer when the atmospheric drag is detected by the science accelerometer. The probe separation sequence is initiated by the aeroshell sequencer discrete that commands BVS separation from the aeroshell. The subsequent BVS and science deployment sequence is initiated by an ambient pressure switch that issues a discrete at a preset pressure level. The BVS sequencer mission is complete when the balloon inflation has been completed. The sequencer is mounted on, and separates with the tank truss. The weight of the sequencer is estimated at 3 lb and requires 2W.

Attitude Control

Attitude control of the entry capsule provides proper orientation for the deorbit thrust and initial angle of attack at entry. Attitude control is terminated at entry. At entry aerodynamic forces and moments control the attitude of the aerodynamically stable configuration.

Spin Stabilization

For the orbit and flyby missions a trajectory has been selected in which the velocity impulse and atmospheric entry attitudes are virtually the same. The spacecraft assumes this attitude before capsule separation eliminating capsule reorientation maneuvers after separation. Spin stabilization has been selected as the attitude control method.

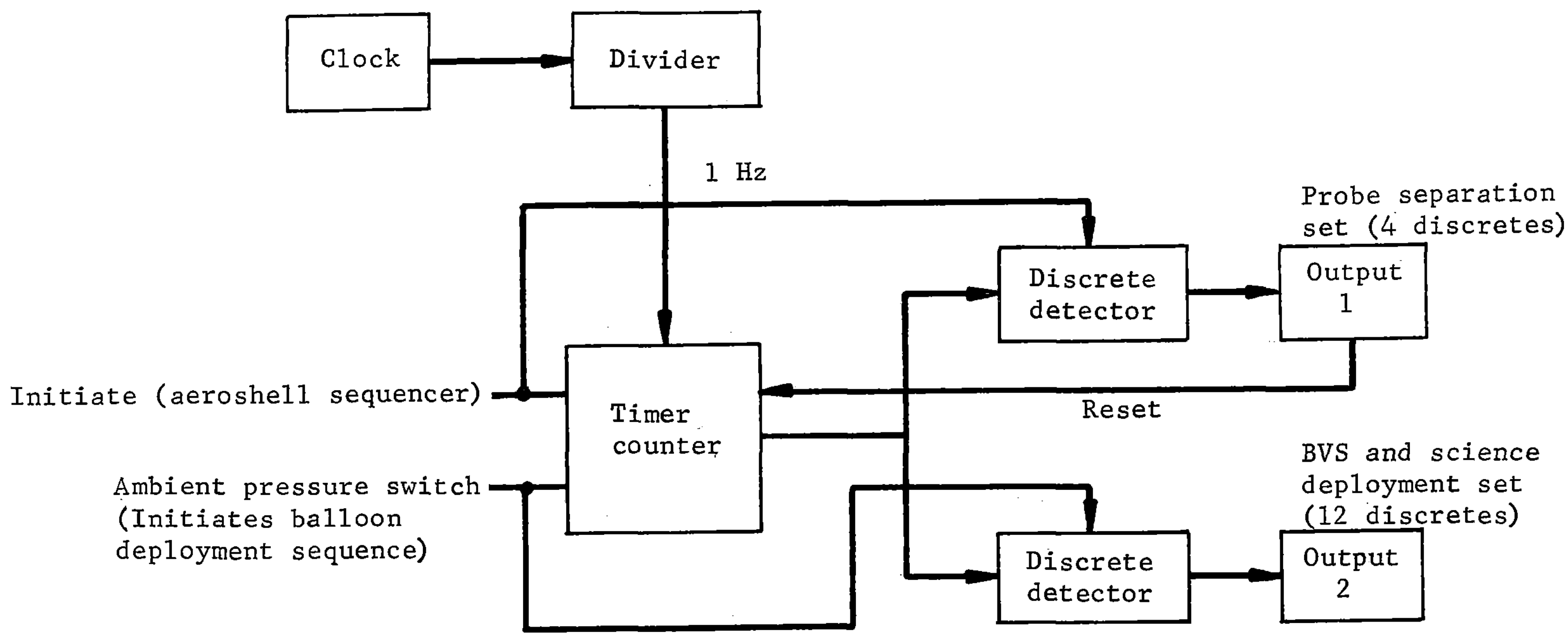


Figure 144.- BVS Sequencer

Stabilization is achieved by establishing an angular momentum vector in space by spinning up the capsule at the desired attitude (separation attitude). The angular momentum vector maintains this attitude unless acted upon by disturbance torques. For these missions, torques due to solar pressure and gravity gradient have negligible effect during coast to the atmosphere. Body torques due to thrust misalignment and offset during engine firing result in nutation of the spin axis and angular momentum vector with an associated pointing error. These disturbance torque effects are controlled by the spin rate.

Other deorbit velocity pointing error sources are tip off rates at separation, coast time, spin up time, and initial nutation rates at deorbit engine ignition.

To minimize angle of attack convergence problems during entry, a partial despin with yo-yos will be performed before atmospheric encounter.

The advantages of using spin stabilization in this application are that it is simple, inexpensive, lightweight, and proved system that will satisfy system requirements. It also requires a negligible amount of electrical power. The system weight is estimated at 8 lb.

For the Venus/Mercury mission the attitude control concept selected consists of a cold gas ACS system for the initial maneuver phase, and spin stabilization for the long coast and entry phase, with a partial despin before atmospheric entry. The system block diagram is shown in figure 145.

Three-axis attitude control provides the required accuracy during the maneuver phase. An inertial measurement unit (IMU), containing three orthogonal single-axis gyros and one pendulous accelerometer along the thrust axis of the capsule, is required to operate from separation to spin-up, approximately 15 minutes. It is initialized by the spacecraft attitude reference equipment and is used to provide and maintain a control reference through the cold gas reaction jets. It is also necessary for orienting the vehicle and maintaining this orientation while firing the deflection engine. The three-axis attitude control system is not used after spin-up.

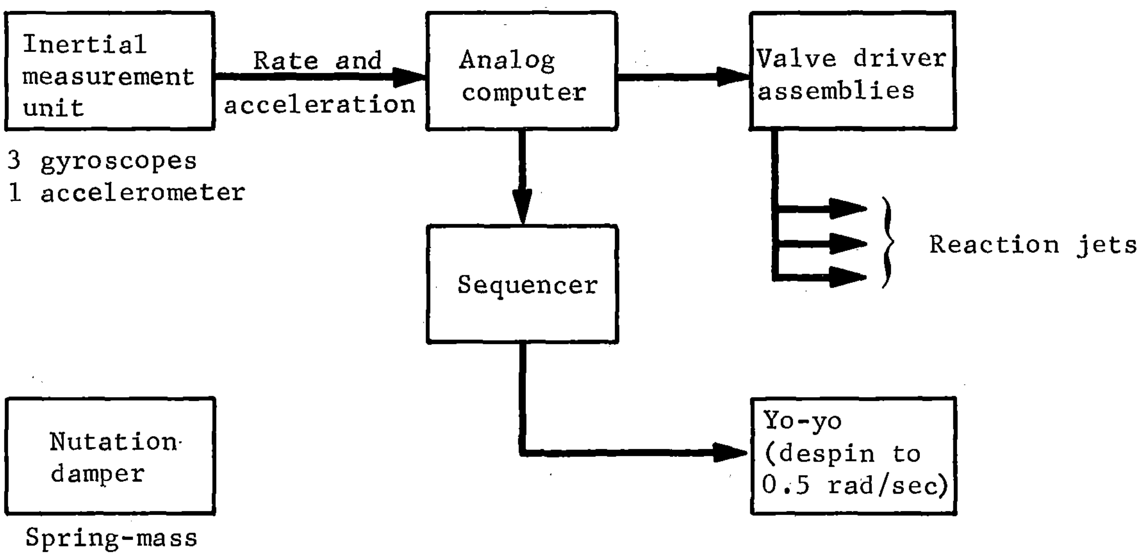


Figure 145.- Guidance System Block Diagram

The cold gas system is sized to produce 35 ft-lb of torque in the roll axis to allow spin up in approximately 2 sec. The pitch and yaw axes jets are sized to produce 2 ft-lb of torque to overcome misalignment of the 50-lb thrust engine. Although the complexity of the three-axis attitude control system is considerably higher than that for spin stabilization, because of the short time in which it is required to operate, reliability is not considered to be significantly affected. Gyroscopic drift of 0.25°/hr is insignificant, and allows for rugged, less accurate gyroscopes to be considered.

Spin stabilization is used during the 68-hr coast phase of the Venus/Mercury mission, because after the deflection velocity reorientation maneuver no capsule reorientation maneuvers are required. In addition, spin stabilization is a simple, lightweight, and reliable approach requiring no electrical power. The system consists of the cold gas reaction jets for spin-up, a nutation damper, and yo-yo mechanism for partial despin from 2.20 rad/sec to 0.5 rad/sec. These reaction jets are those used during the three axis control phase for roll control.

Initial capsule orientation is established by the three-axis attitude control system which is turned off after spin-up. Spin-up is accomplished by firing two of the cold gas jets. Transverse torques during spin-up due to differential thrust, thrust misalignment, and errors in mounting the ACS jets result in nutation or wobbling of the spin axis after spin-up. A nutation damper consisting of a damped spring-mass system is provided to remove these motions. The effects of solar pressure appearing as a constant inertial torque during coast will cause negligible precession of the angular momentum vector.

Due to uncertainties in entry dynamics, heat shield requirements, and environmental requirements, a partial despin before atmospheric entry is included in the configuration. A spin rate of 0.5 rad/sec during entry provides the benefits of rapid angle of attack convergence, and a more stable condition. Partial despin is applied before atmospheric entry to provide adequate stabilization in the absence of stabilizing aerodynamic torques, and thereby assure proper entry attitude. The yo-yo mechanism was selected for this despin operation. The final spin rate of 0.5 rad/sec was selected to maintain sufficient angular momentum so that the torque impulse due to an imperfect yo-yo cable release would not cause an attitude reorientation of greater than 1°. The total ACS system weight is estimated at 48 lb and requires 25 W continuously during the three-axis control period.

OPERATIONAL SUPPORT EQUIPMENT

Requirements and constraints for the operational support equipment (OSE) are dictated by the physical and functional design of the capsule system, by the various incremental assembly level tests required, and by the number and location of test facilities.

The physical and functional design of the capsule, its major modules or segments, and related subsystems, together with their internal and external interfaces, establish the basic OSE requirements for operation and handling of the system. The BVS/entry vehicle (or capsule) system consists of the following major segments as determined by assembly/test considerations:

1) Buoyant Venus station;

2) Inflation module of BVS;

3) Drop sonde (2);

4) Subsonic probe;

5) Aeroshell and afterbody;

6) Deorbit module;

7) Canister/adapter.

Segments 1, 3, and 4 are self-sufficient modules designed to operate within the Venusian atmosphere, and must be assembled and tested accordingly. The remaining segments, when incrementally assembled with the BVS, probe and sondes, provide the deployment, entry, deorbit, and cruise mode capabilities of the system. They require testing at their individual assembly levels and at each higher level of assembly representative of a specific mission phase.

Figure 146 is a flow chart that relates the usage of OSE to various assembly test levels and to the several test facilities needed in support of test operations.

The figure, while not a test flow, is a convenient summary of test flow requirements, and permits classification of OSE by assembly/usage/location. The flow defines assembly test levels at component/subassembly, subsystem, major module, and system. Each of these levels represent logical incremental steps in the development, qualification, and acceptance of the capsule system.

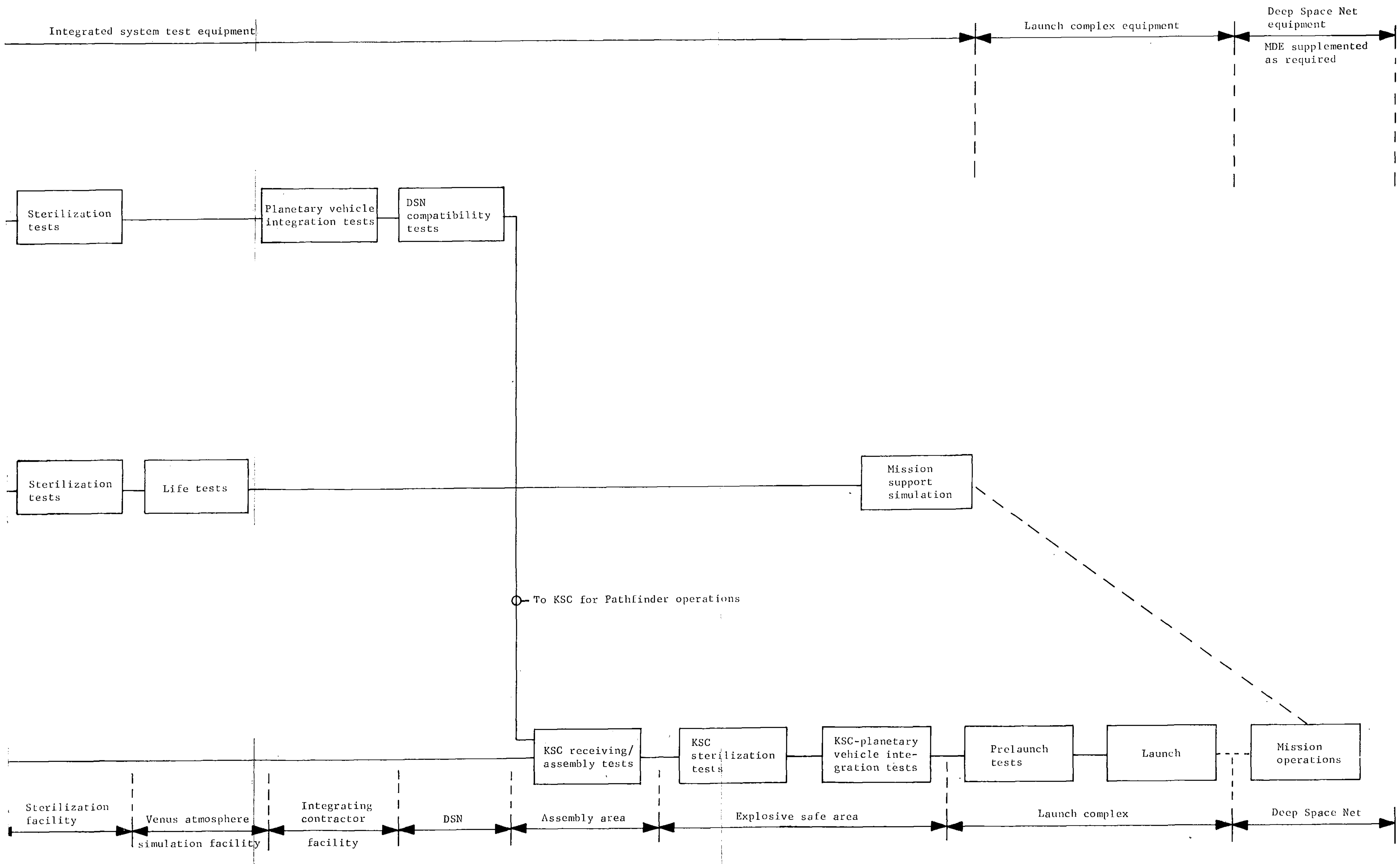


Figure 146.- OSE Usage Requirements, BVS

Each level requires that OSE be provided that can exercise the flight hardware and simulate its interfaces with other system elements. Consequently, OSE test sets and handling equipment can be classified according to the assembly levels that they are designed to support. This results in OSE classifications (identified in figure 146) that correspond with the assembly test levels listed above. In addition, special test models, launch operations, and mission operations establish location-peculiar demands for special test equipment, launch complex equipment (LCE), and mission-dependent equipment (MDE).

Because of parallel or overlapping schedule considerations at, or between, assembly levels, several sets of OSE may be required for each level. To reduce the quantity of OSE required, each assembly test level has been evaluated with respect to its value to the test program. The result of this analysis was to eliminate the subsystem test level beyond the development phase. The reasons for eliminating this test level are:

1) The dispersion of subsystem components/assemblies between various system segments results in the creation of an artificial configuration at a subsystem assembly level;

2) To test at subsystem levels requires considerable functional and environmental simulation which, at best, can only be an approximation of actual conditions;

3) The major module/system test levels provide a more realistic environment for subsystem evaluation.

The subsystem test level has been retained in the development phase as an engineering tool to permit functional evaluation of design concepts and alternatives before integrated system build and operation.

A second cost reduction consideration was an analysis of the need for formal identification and configuration of assembly level test sets. In many instances, at component/subassembly and subsystem levels, the test equipment required is either standard, general-purpose-type equipment, or special-purpose equipment not suitable for use beyond the laboratory development/qualification phase. Furthermore, formal identification of test set configurations at these levels is constraining on development organizations to the extent that time and cost impacts are considerable. The alternative to formal configuration identification and control of test sets was to revert to previous practices that exercised control at qualification/acceptance test levels through engineering and quality control of test procedures and accompanying test tooling configurations.

Adoption of the above two cost reduction features focuses considerable attention on the adequacy of testing at major module and system levels. It also correspondingly sets requirements for implementation of OSE test sets at these levels. It is at these levels that the more realistic operating modes and environments are encountered, and adequate evaluation testing at these levels will yield the most beneficial results.

REQUIREMENTS

The following specific functions will be performed by the OSE:

1) Test control to permit operation of the capsule system in all operational modes;

2) Simulation of the capsule system power sources and spacecraft power interfaces;

3) Stimulation of science experiments;

4) Stimulation of environmental sensors;

5) Interface signal simulation and loading;

6) Simulation of nonreversible functions such as pyrotechnic device operation;

7) Simulation of deployment mechanism operation;

8) Simulation of propulsion system operation;

9) Sequence control in conjunction with, or in lieu of, flight sequencer operation;

10) Variation of input stimuli to permit design limit and marginality testing;

11) Malfunction isolation to replaceable assembly levels;

12) Data acquisition, processing, and display to permit evaluation of capsule system performance during test operations;

13) Commodity supply and servicing;

14) Hazard monitoring and safing control;

15) Alignment, positioning, handling, and transportation.

CONSTRAINTS

Specific constraints imposed upon the capsule system OSE are:

1) Limitations imposed on test interface access to system functions by the physical design of such items as the subsonic probe, drop sondes, aeroshell/afterbody, and the sterilization canister. These limitations are reinforced by test operations during which test access is limited by requirements to maintain test realism, such as in thermal-vacuum and electromagnetic interference testing;

2) Hazardous and/or nonreversible operations such as engine firing and pyrotechnic device ignition which are restricted to subsystem tests in hazard-safe areas. These functions also constrain poststerilization operating tests after installation of propellants and pyrotechnic devices;

3) Progressive testing at major module and integrated system levels which requires programing and interface flexibility within the OSE to permit common usage at various test levels;

4) The number of required test and operations facilities and their locations that require provisions within the OSE for expeditious relocation and setup of the support equipment;

5) Integrated prelaunch and launch operations that require interface compatibility with other elements of the space vehicle and facilities;

6) Maximum use of DSN mission independent equipment (MIE) for mission operations.

CONFIGURATION DESCRIPTION

In figure 146 the following OSE test and support equipment classifications are identified:

1) Component test equipment;

2) Subsystem test equipment;

3) Special test equipment;

4) Major module test equipment;

5) System test equipment;

6) Launch complex equipment;

7) Mission-dependent equipment.

This report concentrates on the equipment configurations for support of testing at major module level and above. It should be recognized, however, that a portion of the system OSE design will evolve from designs developed at component and subsystem levels, particularly for the more unique equipment items.

The development of a configuration for major module and system assembly levels was subject of a study involving several possible alternatives. The selected configuration appears to provide the most versatility in supporting system operations.

Realistically, the choice of OSE configuration cannot be made based specifically on requirements dictated by any single assembly test level. In the selection of a configuration, requirements at major module, system, and launch complex levels of test or operations were considered and an OSE configuration was selected that would best satisfy the total usage requirements. The system proposed is based on the use of a centralized general-purpose digital computer to achieve the versatility needed to support several different assembly test levels in a number of different test locations. The use of a centralized computer also provides a real-time and near real-time data processing capability. Elements of the software system provided to support data processing functions can be directly applied to mission operations support, thus extending the common usage aspects of the system.

SYSTEM TEST COMPLEX

The configuration is shown in figure 147 and is identified as the BVS/Entry System-System Test Complex. It consists of a general-purpose digital computer system located in a control center area remote from the test area, and a group of unique test and support equipment and computer interface hardware located in the vicinity of the capsule. The system is patterned after an existing computer system design used in the Titan IIIM-MOL program. This system contains a third-generation, 32-bit, medium-size computer that interfaces through general-purpose checkout equipment to the test article. It is designed for remote operation of up to 2000 ft via biphase transmission links, and is ideally suited for use in test and operations areas where hazardous conditions restrict personnel access. The use of an existing design eliminates many of the development problems associated with a new design and permits the carryover of basic software programing and experience to the new application.

The computer system performs test control, and real-time and near real-time data processing. Input-output peripheral equipment is provided to support data display requirements and the preparation of detailed test records. Adequate storage for on-line test programs is provided in core-memory and rapid access disc files. Off-line programs are supplied for history file preparation, test-to-test trend comparisons, and miscellaneous data reduction and computation tasks. Computer interface equipment at the control center receives computer commands, and formats and verifies the command instruction before transmission over the biphase transmission links. Data acquired in the test area are received and preprocessed in computer interface equipment provided for that purpose.

Capsule-vicinity OSE is used to convert computer commands into bilevel, analog, or digital signals for stimulating capsule operations or simulating functional interfaces with other system elements. This equipment is general-purpose and signal channels, amplitudes, frequencies, wave shapes, etc., are programed by computer instruction. Telemetry, bilevel, analog, and digital data is acquired, formatted, and transmitted to the Control Center for processing and display. Some capsule-vicinity OSE is of necessity unique to the BVS/entry system configuration. RF transmitters/receivers, science experiment stimuli and monitoring, hazard monitoring and control, propellant loading and pressurization, solar panel checkout, etc., all require special equipment to complete system checkout capabilities. Where used in automatic system checkout routines, this equipment is programed and controlled by the computer system.

Where practical, capsule-vicinity OSE is van-mounted to simplify teardown, setup, and checkout when equipment is moved between test facilities. The use of a centralized computer system reduces the amount of capsule-vicinity OSE that must be moved between test areas.

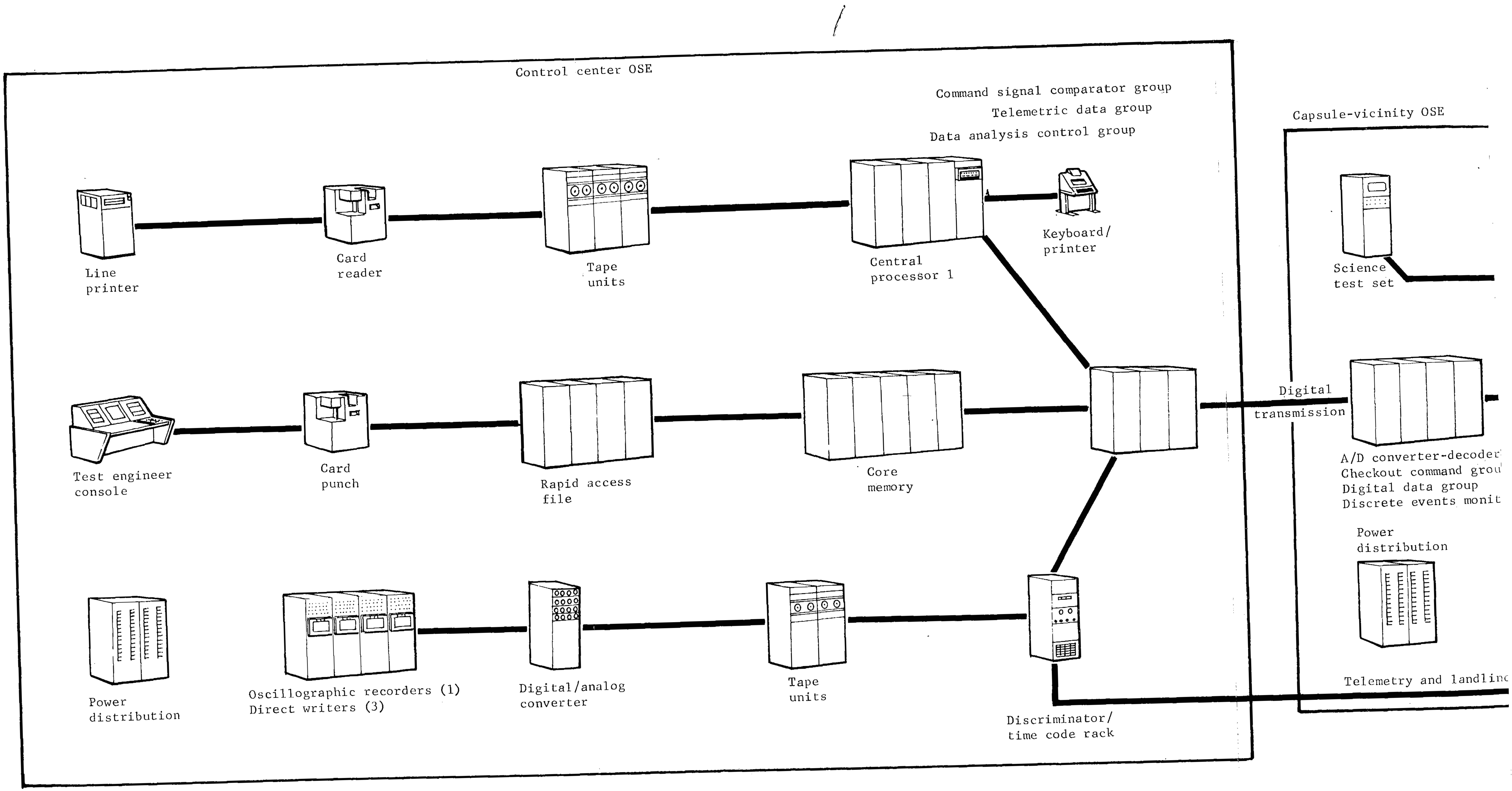
The proposed configuration consists of four capsule-vicinity vans, one for each capsule. These vans are connected to the central computer system by transmission links provided to each test area. Two computer sets are to be supplied, both of which will initially be used at the contractor's facility for system development and qualification. The two computer sets will be time-shared with the four van sets as test needs require. As off-site usage dictates, one computer set will be relocated with the necessary van set or sets. Ultimately, one computer set will be located at the contractor's facility and the other at KSC.

Capsule-vicinity OSE that is not van mounted is not generally provided on a one-for-one basis with the capsule systems. For example, since sterilizable batteries, solar panels, propulsion, and balloon inflation systems cannot normally be exercised in routine testing of the system, support items for these functions can be more effectively time-shared and equipment quantities reduced accordingly.

The system test complex contains both the general-purpose capabilities and the unique equipment necessary to support testing at either major module or system assembly test levels. In most instances, the general-purpose capabilities of the system will adequately support differences in interface configuration between the separate test levels. While there may be some unique items required at module level over and above those needed at system level, they are of negligible proportion with respect to total configuration requirements.

SYSTEM TEST COMPLEX OPERATIONS

The system complex operates in the following manner. All programs, including system monitor programs, test and data acquisition and processing routines, and success criteria and data tables are entered into computer memory via the card reader or the magnetic tape units.



Control center OSE
Command signal comparator group
Telemetric data group
Data analysis control group
Line printer
Card reader
Tape units
Central processor 1
Keyboard/ printer
Test engineer console
Card punch
Rapid access file
Core memory
Power distribution
Oscillographic recorders (1)
Direct writers (3)
Digital/analog converter
Tape units
Discriminator/ time code rack
Digital transmission
Capsule-vicinity OSE
Science test set
A/D converter-decoder
Checkout command grou
Digital data group
Discrete events monit
Power distribution
Telemetry and landlinc

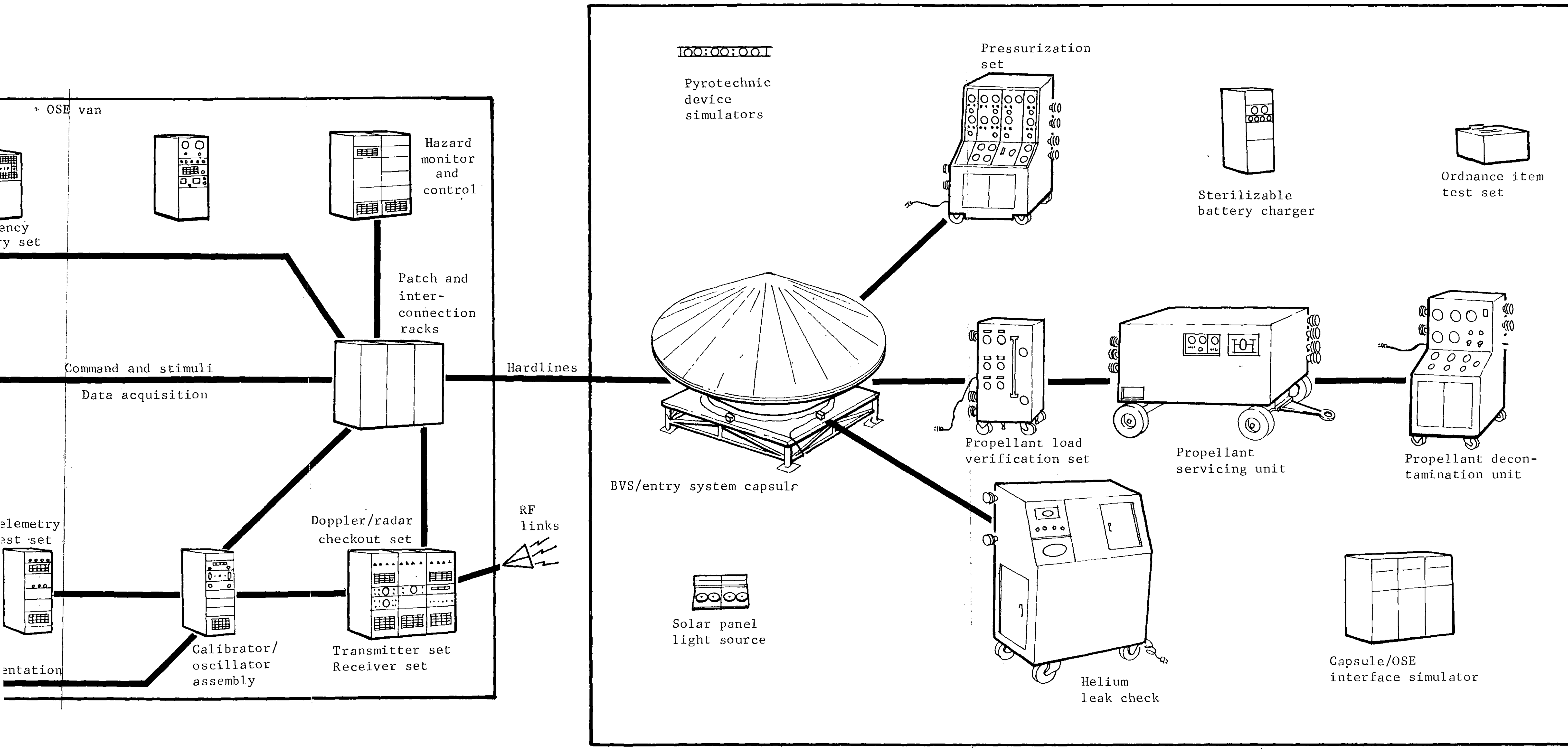


Figure 147.- BVS/Entry System - System Test Complex (Computer System)

The test conductor monitors activation and status of OSE systems from the test engineer console. A keyboard-printer is provided for test engineer communication with the computer system. This unit can be used to schedule computer input/output operations, to modify test sequences, to update success criteria and data tables, and to call up test routines or data values for printed display. The test engineer console contains test controls, status monitors, and voice communications. Hazard monitors and safing controls are prominently located on the console panel.

After satisfactory system status has been assured, the test engineer initiates the test sequence by commanding computer operation. On initiation, the computer will automatically sequence the test as defined by the test routine programed into its memory. The computer first activates prerequisite OSE functions and verifies their status, after which, it commands the operation of the capsule system sequencer. The sequencer then controls capsule system functions as programed for actual mission operations. The OSE computer is synchronized with the sequencer and commands the application of simulated response and stimulus signals at appropriate times in the sequence.

The telemetry subsystem is used as the prime test data source. Telemetry data are transmitted open-loop via the communication subsystem to the communication OSE where it is received, demodulated, and passed on to the telemetry OSE. As an alternative, the communication OSE may be bypassed and telemetry data hardlined directly to the telemetry OSE. The telemetric data group (TDG) establishes bit synchronization, evaluates signal quality, and provides a reconstituted PCM serial data stream. The TDG, under control of preprogramed computer software routines, identifies format, decommutates and decodes the data, and performs limit checks and success criteria comparisons. Data failing the limit check or criteria comparison are printed out on the line printer. Event data indicative of test progress and status are also printed and selected signals displayed on the test engineer console. These displays permit operation identification of system anomalies and assist test control decisions.

The system test complex will provide simulated sequencer and telemetry clock signals so that sequence timing and data acquisition rates can be varied as required by test operations.

Telemetry data are supplemented by discrete and analog data monitor systems. The discrete monitor system detects changes in discrete signal levels and transmits the change data to the computer for criteria comparison. Data failing criteria checks are displayed as alarm data on the line printer. The analog monitor system converts and formats capsule hardlined analog data for processing by the computer and recording of selected channels on strip-chart recorders. These recorders may also be driven by the computer through digital-to-analog converters to display analog data available on telemetry channels.

The discrete command and analog stimulus system accepts digital commands from the computer, decodes the digital address, and selects the appropriate output signal for application to the capsule system interface. Programable analog stimulus generators are provided. Analog stimulus levels are preselected before start of the test sequence. They are applied to the appropriate interface by computer command through discrete output switching.

During system testing, backup data recording is provided by analog tape recorder equipment that accepts predetection rf signals and other multiplexed data signals. Time and voice signal channels are also provided. This equipment permits reconstruction of test data for off-line analyses. The high frequency capability of this equipment is also used to detect and analyze transient and electromagnetic interference (EMI) phenomena.

To minimize facility interfaces and to permit rapid relocation and setup of the system test complex, a power distribution group is provided to distribute ac and dc power to all system test complex end-items. This power distribution group contains dc power supplies, distribution buses, and power monitoring and switching equipment.

Unique capsule-vicinity OSE is used in test operations to support the following functions.

The power control and monitor rack provides power to the capsule system buses in lieu of capsule battery sources. It also simulates spacecraft power. Power application and transfer control is effected automatically from the computer or manually through power rack controls. Capsule power metering and load control functions are also included in this rack.

The hazard monitor and control set consists of a propellant and pressurization monitor rack, and a pyrotechnic subsystem rack. These racks are configured and grouped to permit common usage in capsule storage areas and at the launch pad for system and personnel safety.

The propellant and pressurization monitor rack monitors temperature and pressure and vent controls for propulsion and inflation system functions.

Safe monitoring and control of pyrotechnic subsystem safe/arm devices are provided by the pyrotechnic subsystem rack. It also monitors power capacitor banks and provides control of power application and discharge of these assemblies.

Communication OSE through receiver/transmitter sets receives signals from the capsule and demodulates these signals to recover the telemetry data signal. RF carrier frequencies are verified and spectrum analysis tests performed. Capability also exists to generate test patterns, to measure transmitter input/output power, and to detect bit errors in demodulated telemetry signals. A transponder is provided for transmission of return signals to the radar altimeter. Doppler simulation is provided for verification of capsule uhf doppler measurement. These latter functions are contained in the doppler/radar checkout set.

A telemetry test set is provided to view and analyze the quality of telemetry transmissions. Telemetry mode control is also performed from this equipment.

The propellant loading and pressurization equipment is used to load actual or simulated propellants and to pressurize propellant systems and balloon inflation systems. It consists of two types of propellant servicing units, a decontamination set, a propellant load verification set, and a pressurization set. A helium leak check set is used to detect leakage in system plumbing.

A solar panel light source is used to assure the continued integrity of solar panels after assembly on the BVS.

After initial setup of the system test complex, the capsule/OSE interface simulator is used to verify OSE operation before marriage of the OSE with the capsule system. This self-test of the OSE is performed after each relocation and whenever OSE operation is suspect. Additional self-test features built into system test complex equipment verify interface signals and permit isolation of malfunctions between the system test complex and the capsule system.

Functional test sequences require the use of a pyrotechnic device simulator to substitute for pyrotechnic devices which, because of their hazardous, nonreversible characteristics, cannot be used routinely in test operations. Pyrotechnic device simulators are provided to simulate the firing of pyrotechnic devices and to measure energy levels applied through the firing circuitry.

During component evaluation tests and before installation of active pyrotechnics in the capsule, tests are conducted to assure continuity, rf immunity, and ground isolation of squib circuits within pyrotechnic devices. This testing is performed with equipment provided in the ordnance item test set.

After loading of propellants and pressurants, and installation of sterilizable batteries and pyrotechnic devices, potentially hazardous conditions exist within the capsule system. To assure continued operation of the hazard monitor and control set in the event of prime power failure, an emergency battery set is provided to supply nominal 28 Vdc power to capsule transducers and ground monitoring equipment.

Flight batteries are normally formation charged at the manufacturer's facility before delivery for system installation. However, to assure proper battery operation after exposure to sterilization temperatures, present concepts require battery formation charging after sterilization. A sterilizable battery charger set is provided to perform formation charge/discharge cycling.

The specific operation and configuration of the system test complex will vary with each test operation, depending on test objectives and interface availability. The general-purpose computer operating through addressable switching matrices and patching networks will provide the flexibility needed to support these varied operations.

At the contractor's plant, the system test complex is required to support operations in the spacecraft assembly and test area, the Venus atmospheric simulation facility, the dynamics test facility, and the thermal vacuum test facility. The system test complex computer system will be located in the spacecraft assembly and test area and will interface the mobile capsule-vicinity van units in each facility area via data transmission links.

At KSC, a similar facility support requirement exists. Operations in an assembly area, an explosive safe area, and at the launch complex will be supported by the system test complex computer through appropriate data links. Within the explosive safe area itself, several test areas are needed to support pre- and poststerilization testing and planetary vehicle integration. The STC computer is assumed to be located in the explosive safe area because of the greater number of operations to be conducted at this facility.

Because of the severity of the sterilization environment, poststerilization functional testing is considered mandatory. The configuration of the capsule at this time, however, imposes operating and access constraints because of the presence of active pyrotechnic devices and the sealed sterilization canister. While it will be necessary to provide hardline umbilical access through the canister for power and hazard monitor and control functions, it is desirable to limit such access to an absolute minimum. It will be advantageous, therefore, to provide some built-in test features into the capsule to facilitate poststerilization testing. These test provisions will include:

1) Ability to isolate and disable sequencer commands for pyrotechnic initiation;

2) Single-point calibration stimuli for science experiments;

3) A means for positively verifying the removal of command disables.

During this test, almost complete dependence is placed on required telemetry data for test evaluation. The availability of supplemental discrete, analog, and digital data via hardline interfaces is virtually nonexistent. Poststerilization testing therefore establishes certain test philosophies and constraints that carry back into previous design, development, and qualification activities in order to assure equipment capabilities and success criteria values for use in these tests.

LAUNCH COMPLEX CONFIGURATION

The configuration of the launch complex equipment is shown in figure 148. The configuration proposed is essentially a system test complex with a minimum amount of capsule-vicinity OSE. The centralized system test complex computer is used to support launch operations after completion of KSC final assembly, sterilization, and planetary vehicle testing. This computer remains in the same area as for previous tests and is connected to the launch complex by rf or landline data links. At the launch complex, only power, hazard monitor and control, and H_2 pressurization equipment is required. An emergency power supply is provided to assure continuity of power in the event of facility power outages.

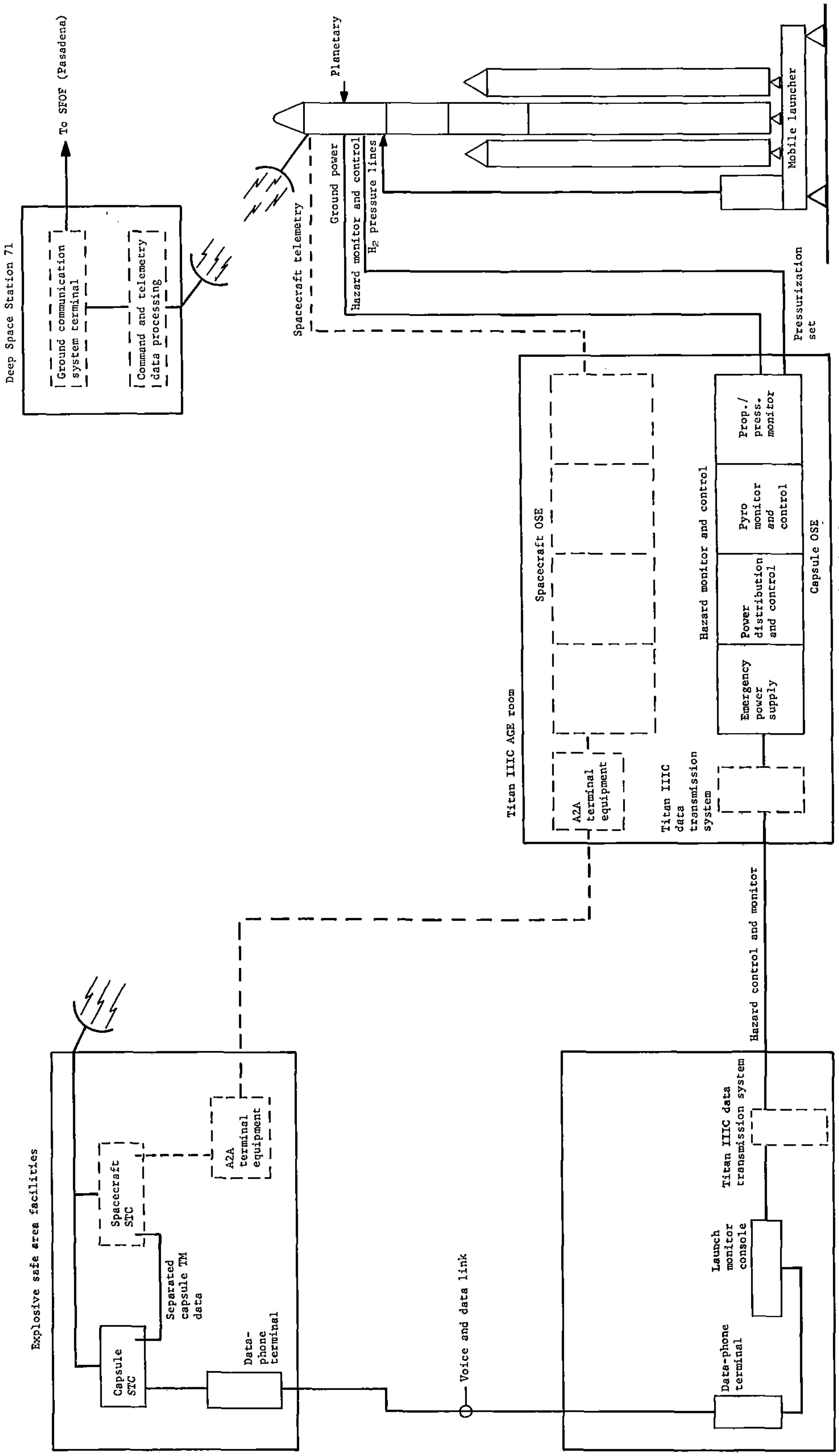


Figure 148.- Launch Complex Equipment

In the launch control center, a launch monitor console is provided that interfaces with both launch pad equipment and the system test complex computer system. Hardline interfaces with the launch pad are routed over the Titan IIIC data transmission system. A data-phone link is provided between the launch control center and the STC computer.

Operation

At launch, the BVS capsule is in a passive state. The only active equipment is that required to provide a status monitor function. The spacecraft provides instrumentation power and data telemetering. During prelaunch checkout, test operations are constrained by the presence of onboard propellants and pyrotechnics. Checkout is therefore limited to whatever status information is available from the capsule in a nonoperating status. Nonoperating status in this context implies the powering up of the capsule subsystems without, however, initiating any sequenced operations.

Because the spacecraft combines capsule telemetry data with its own during mission operations, it is assumed that open loop and/or landline transmission of capsule telemetry data will be via spacecraft equipment and data transmission channels. After data separation by the spacecraft OSE, capsule telemetry data are routed to the capsule system test complex computer for processing and evaluation. Capsule status signals are then transmitted via data-phone link to the launch control center where they are displayed on the launch monitor console. This data link also provides voice communication between the launch control center and system test complex operators.

Continuous monitoring of potentially hazardous functions is provided by the hazard monitor and control equipment at the launch pad. Propellant and pressurization tank temperatures and pressures, and pyrotechnic device safe/arm monitors constitute the majority of these functions. Manual safing controls are provided both at the launch pad and from the launch control center. All hazard monitors and controls are via direct hardlined interfaces with the capsule. The Titan IIIC data transmission system is used to provide a highly reliable interface between the launch control center and the launch pad.

Power is supplied to the capsule to permit capsule activation without depleting capsule battery power supplies. These supplies are located in the power control and distribution rack, which also contains capsule power transfer and load control provisions. Power is also distributed to other launch control equipment with an emergency power supply providing backup power to assure continuous monitoring and safing capabilities.

The capsule incorporates provisions for sterile loading of hydrogen gas for the balloon inflation system. A pressurization set is provided to pressurize inflation system tanks. The system is sealed by pyrotechnically operated valves and tubing cutters controlled from the hazard monitor and control set.

During the boost and orbital insertion phases of the mission, command and telemetry interfaces are established between the spacecraft and Deep Space Station 71. Spacecraft and capsule status monitor data can therefore be monitored at the Mission Operations Center in the Space Flight Operations Facility at Pasadena by data links provided by the ground communication system. If a line-of-sight repeater antenna system is provided as part of the spacecraft launch support system, this same capability can be achieved during prelaunch and launch operations.

MISSION-DEPENDENT EQUIPMENT (MDE)

The function of MDE is to supplement the general-purpose mission independent capabilities of the Deep Space Net (DSN) during mission operations. The DSN provides capability for command control; data acquisition, processing, and display; and, range and rate tracking. MDE is only required where special data encoding or data display needs exceed DSN capabilities. For the Venus mission, all capsule data and command messages are relayed via the orbiting spacecraft and the capsule therefore has no direct interface with the DSN. The DSN does decode, process, and display separated capsule data. No special coding or display requirements are presently projected that would require the use of MDE and the general-purpose capabilities of the DSN should be sufficient.

The spacecraft, because of the addition of orbital science, may require MDE to support display and display processing of data, such as for photo-imaging experiments.

MISSION-DEPENDENT SOFTWARE (MDS)

General-purpose mission independent equipment must be supported by mission-peculiar software, particularly in the areas of telemetry data decommutation and processing. The capsule MDS effort will be to supply software subroutines for integration into DSN operational programs. These subroutines will provide the procedures necessary to decommutate capsule data formats and to perform data processing functions. These software subroutines will be derived from software programs used in support of development, qualification, and acceptance testing, and will therefore have been proved by actual use before their incorporation in mission operation support. Coordination with DSN software organizations will be required because of possible differences in coding that may exist if computer types are different, and to assure proper integration and interfacing with DSN operational system programs.

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT (AHSE)

This equipment category includes those items of assembly, handling, servicing, shipping, and transportation equipment which is supplied as capsule system-peculiar equipment. It includes assembly stands, positioning fixtures, packing cases, alignment equipment, special tools, cleaning equipment, transporters, canister pressurization, and conditioning equipment, etc.

In addition to the AHSE, facility common usage equipment such as cranes, hoists, forklifts, trucks, etc., will be used in support of assembly and test operations.

FACILITIES

At the contractor's plant, facilities needed in support of assembly and test operations are provided by the contractor. In addition to general building structures and services, test equipment is provided to perform vibration, shock, acoustic, EMI, thermal-vacuum, sterilization, and Venus environmental tests. Gas and propellant storage facilities, contaminated propellant disposal facilities, test instrumentation, communications, power, etc., used in direct support of test operations is also contractor provided.

At KSC, it is assumed that government-furnished facilities will be provided that are modified to contractor requirements for assembly and test operation support.

OSE CONFIGURATION DIFFERENCES, FLYBY VS ORBITAL MISSION

Capsule system differences for the 1972 flyby mission and the 1973 orbital mission are as follows:

1) Deflection propulsion is a monopropellant system rather than the 1973 orbiter bipropellant system;

2) Power subsystem solar panels are deleted from this configuration;

3) An S-band direct communication link to the DSN is provided to support command and data return functions after entry and deployment phases, which continue to use the uhf relay link to the spacecraft.

Changes to the basic 1973 orbiter mission OSE to support a 1972 flyby mission are described in the following paragraphs.

System Test Equipment

Capsule system differences are not significant enough to alter the results of the system test complex configuration study. The earlier mission data emphasizes the schedule availability factor and reinforces the need for application of a system of primarily existing design. The central computer system is therefore retained for this mission.

Differences in the system test complex equipment complement are:

1) Only one propellant servicing unit is required for the monopropellant system as compared with the two types needed in support of a bipropellant system;

2) The solar panel light source is deleted;

3) S-band capabilities are added to the transmitter/receiver sets to support checkout of the S-band direct link communications on the capsule.

Other differences are the reduction in the number of pyrotechnic system functions, the deletion of solar power regulators, and changes in the number of control and monitor signal interfaces with the capsule. Because of the general-purpose nature of the OSE, these differences have very little impact on the 1973 orbital mission baseline configuration.

Launch Complex Equipment (LCE)

No differences exist in LCE for this mission configuration.

Mission Dependent Equipment (MDE)

For the orbital mission there was no direct interface with the DSN. The flyby mission, however, adds an S-band direct-link for postdeployment communications. This interface imposes a requirement on the DSN for support of both spacecraft and BVS S-band links. This capability exists within the DSN, however, and no special MDE is needed.

Mission Dependent Software (MDS)

The flyby mission will add somewhat to the minimum MDS effort defined for the orbital mission. Because the DSN receives postdeployment data direct from the capsule, software programing to establish bit rate and frame synchronization must be provided. Software will also be provided for formatting and coding ground commands to the BVS.

OSE CONFIGURATION DIFFERENCES, SWINGBY VS ORBITAL MISSION

Capsule system differences for the 1973 Venus/Mercury swingby mission from the 1973 orbital mission are as follows:

1) Deflection propulsion is a monopropellant system rather than the orbiter bipropellant system;

2) Power subsystem solar panel are deleted from this configuration;

3) S-band direct communication links to the DSN are provided for both the BVS and the subsonic probe instead of the uhf relay links to the spacecraft;

4) An active attitude control system is provided for this mission in place of the spin stabilization system provided for the orbital mission. The active ACS will be a cold gas system consisting of reaction jets, solenoid control valves, valve drivers, a small digital computer, and an inertial measurement unit.

Changes to the basic orbiter mission OSE to support a Venus/ Mercury swingby mission are described in the following paragraphs.

System Test Equipment

Capsule system differences are not significant enough to alter the results of the system test complex configuration study. The central computer system is therefore retained for this mission.

Differences in the system test complex equipment complement are:

1) Only one propellant servicing unit is required for the monopropellant system as compared with two types needed to support a bipropellant system;

2) The solar panel light source is deleted;

3) S-band capabilities are added to the transmitter/ receiver sets to support checkout of the S-band direct-link communications on the BVS and the subsonic probe. UHF capabilities are not needed and can be deleted;

4) The G&C subsystem change will require that the computer, inertial measurement unit, solenoid valve drive signals, and solenoid operation be verified during system tests. These checkout requirements, however, are within the general-purpose discrete, analog, and digital capabilities of the system test complex.

Launch Complex Equipment (LCE)

At the launch complex, provisions will be required to support S-band direct communications. These provisions will include the addition of an rf relay antenna at the launch pad and S-band transmitter/receiver sets in the central computer area. This will permit direct reception of capsule data, which is no longer interleaved with spacecraft data except for a limited amount of status monitor data needed during launch and cruise phases.

Digital interface equipment will also be required at the launch pad to provide G&C computer load, verification, and update capabilities. This equipment will be controlled from the central computer, which will provide required programing over data transmission links to the pad.

Mission-Dependent Equipment (MDE)

The Venus/Mercury swingby mission imposes a requirement on the DSN to support direct communications with the BVS, subsonic probe, and the spacecraft. This capability exists within the DSN, however, and no special MDE support is needed.

Mission-Dependent Software (MDS)

The Venus/Mercury swingby mission will add to the minimum MDS effort defined for the orbital mission. Because the DSN receives postseparation data direct from the capsule, software programing to establish bit rate and frame synchronization must be provided. Software will also be provided for formatting and coding ground commands to the BVS.

CONFIGURATION ALTERNATIVES

Three separate configurations were identified and evaluated for applicability to test and operations for the BVS system at and above the module level. These configurations are discussed in the following subsections.

Module-Level Checkout System

In this configuration test sets are designed specifically to support the module level of assembly and test. To achieve system level test capabilities, the module-level test sets are assembled together to form a system-level test set. Interfacing of these test sets is not a significant problem because of the functional independence of several of the modules, and the feasibility of sequential rather than parallel operation in mission simulation sequences.

Integrated System Test Complex

This test complex is designed specifically to support a system assembly level of test. As such, it contains all the functional test capabilities needed to also support module-level tests. Because of the functional independence of several of the modules, the use of an integrated test set to support module-level tests presents no interfacing problems of any consequence. General-purpose capabilities provided for system-level test are more than adequate to support module-level tests, including the additional interface signal simulation needed at the module level.

The most significant difference between the module-level checkout system and the integrated system test complex is in the amount of provisioned hardware required. Module-level test sets must each contain self-sufficient provisions for test control and data acquisition. In many instances, functional redundancy also exists between test sets to satisfy similar or identical functions within the system modules. These functional redundancies exist particularly in the areas of power, pyrotechnics, sequencing, and communications.

Conversely, however, the module-level test set concept provides considerably more freedom of use, and does not require the exacting time-sharing schedule that the integrated system test complex configuration requires.

Centralized Computer System

A medium sized computer system is located in a central control area and connected to capsule-vicinity interface equipment by data transmission links. Each test area has a data link terminal to which a capsule-vicinity van set is connected for interface with the computer to support test operations in that area.

To support parallel contractor facility and KSC operations, two computer sets are required. Four capsule-vicinity van sets are provisioned. This concept is an extension of the integrated system test complex. It retains the time-sharing concept of the integrated system test complex and centralizes the test control and data display functions. It adds a computer data processing capability not provided in the other configurations, and reduces the amount of OSE that must be relocated between test areas. It also provides remote operation capabilities that are useful for hazardous operations in explosive-safe and launch areas. This system is based upon an existing Titan IIIM design which reduces design and development activities and improves schedule availability. A fourth configuration alternative exists that would provide subsystem-oriented test sets. The subsystem-oriented concept, however, is not compatible with the practical aspects of module assembly and test because many individual subsystem elements are dispersed among the several system modules. This configuration was therefore discarded after preliminary consideration.

Two additional implementation alternatives could be factored into each of the three basic configuration alternatives. These implementation alternatives consider the use of OSE as either vehicle or facility

Facility-oriented OSE would be installed as fixed equipment in the several test areas, while vehicle-oriented OSE would be relocated with the vehicle as it moved from test facility to test facility.

CONCLUSIONS

A 400-lb class buoyant station is feasible with, and produces an attractive addition to, a 1972 flyby, 1973 orbital, or Venus/Mercury mission using a Titan IIIC launch vehicle and either a Mariner or Lunar Orbiter class spacecraft. This class BVS permits a 175-lb gondola with approximately 60-lb allocated to scientific instruments. The remainder is allocated to telecommunications, either uhf relay to the companion spacecraft or S-band direct to Earth, and power supplied by batteries with the option of adding a solar array.

The 1972 flyby and 1973 orbital missions allow the capsule to enter on the light side of Venus. The Venus/Mercury mission requires a dark-side entry. The flyby and orbital mission communications have a uhf link between the BVS and spacecraft for the separation, coast, atmospheric entry, and deployment. The orbital mission continues to use the uhf link to the orbiting spacecraft each orbital pass. The flyby mission switches to an S-band link directly to Earth for the flotation mission. The Venus/Mercury mission geometry requires the BVS to use an S-band link to Earth for all phases of the BVS mission.

Immediate steps toward the development of a heat shield -- depending on the mission selected -- should be undertaken. This is, of course, independent of the desirability of the BVS concept.

The more detailed simulation of the deployment and inflation of the balloon performed in this study and the preliminary design work undertaken has demonstrated that the problem does not represent a significant development risk and is well within today's technology. Some urgency is felt, however, for balloon materials selection and testing effort. Data on the behavior of candidate materials in the required environments are generally incomplete. The demonstration of a full-scale balloon system is a logical early step and will not present significant difficulties because of the ease with which the depolyment and inflation environment can be simulated.

Martin Marietta Corporation
Denver, Colorado, January 8, 1969

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